# NASA CONTRACTOR REPORT

NASA CR-2040



# ATMOSPHERIC RENDEZVOUS FEASIBILITY STUDY

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JUNE 1972

| 1. Report No. | 2. Government Accession No. | 3. Recipient's Catalog No. |
|---|---|---|
| NASA CR-2040 | | |
| 4. Title and Subtitle: Atmospheric Rendezvous Feasibility Study | | 5. Report Date: June 1972 |
| | | 6. Performing Organization Code |
| 7. Author(s): A. D. Schaezler | | 8. Performing Organization Report No.: 00.1481 |
| | | 10. Work Unit No.: 110-06-05-02 |
| 9. Performing Organization Name and Address: Vought Missiles and Space Company, LTV Aerospace Corporation, Dallas, Texas 75222 | | 11. Contract or Grant No.: NAS1-9994 DSI No. 23 |
| | | 13. Type of Report and Period Covered: Contractor Report |
| 12. Sponsoring Agency Name and Address: National Aeronautics and Space Administration, Washington, D.C. 20546 | | 14. Sponsoring Agency Code |

15. Supplementary Notes

16. Abstract

A study was carried out to determine the feasibility of using atmospheric rendezvous to increase the efficiency of space transportation and to determine the most effective implementation. It is concluded that atmospheric rendezvous is feasible and can be utilized in a space transportation system to reduce size of the orbiter vehicle, provide a powered landing with go-around capability for every mission, and achieve lateral range performance that exceeds requirements. A significantly lighter booster and reduced launch fuel requirements are additional benefits that can be realized with a system that includes a large subsonic airplane for recovery of the orbiter. Additional reduction in booster size is possible if the airplane is designed for recovery of the booster by towing. An airplane about the size of the C-5A is required.

| 17. Key Words (Suggested by Author(s)) | 18. Distribution Statement |
|---|---|
| Rendezvous<br>Atmospheric rendezvous<br>Orbital logistics | Unclassified - unlimited |

| 19. Security Classif. (of this report) | 20. Security Classif. (of this page) | 21. No. of Pages | 22. Price* |
|---|---|---|---|
| Unclassified | Unclassified | 114 | $3.00 |

*For sale by the National Technical Information Service, Springfield, Virginia 22151

## FOREWORD

This final report for the "Atmospheric Rendezvous Feasibility Study" completes a study begun 24 May 1971, conducted under Contract NAS1-9994, DSI No. 23, at the request of the Flight Dynamics and Control Division of the Langley Research Center. The purpose of the study is to determine the feasibility of using atmospheric rendezvous to increase efficiency of orbital operations and space transportation and to determine the most effective implementation of the atmospheric rendezvous mode. Preliminary results of the study were presented to NASA at the Langley Research Center on 15 September 1971. Copies of charts used in that presentation are contained in LTV Report No. 00.1472, "Atmospheric Rendezvous Study Review", 15 September 1971. These results are presented and discussed in more detail in this final report. This work was carried out within Advanced Space Systems of Vought Missiles and Space Company by:

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CONTENTS


| | Page No. |
|---|---|
| FOREWORD | iii |
| LIST OF FIGURES | vii |
| LIST OF TABLES | ix |
| SYMBOLS AND ABBREVIATIONS | xi |
| SUMMARY | 1 |
| 1.0 INTRODUCTION | 3 |
| 1.1 Historical Background | 3 |
| 1.2 Objectives | 9 |
| 1.3 Major Assumptions | 12 |
| 2.0 BASELINE CONFIGURATIONS | 15 |
| 2.1 Orbiter | 15 |
| 2.2 Recovery Airplane | 18 |
| 2.3 Boosters | 18 |
| 3.0 RECOVERY CONCEPTS | 25 |
| 4.0 RECOVERY SYSTEM CONCEPTUAL DESIGNS | 27 |
| 4.1 Docking | 27 |
| 4.1.1 Orbiter-Airplane | 27 |
| 4.1.2 Orbiter-Booster | 35 |
| 4.1.3 Docking Dynamics | 38 |
| 4.1.4 Docking Forces | 38 |
| 4.1.5 Effects of Air Turbulence | 40 |
| 4.1.6 Aerodynamic Heating | 40 |

LIST OF FIGURES (Cont.)


| Figure No. | Title | Page No. |
| --- | --- | --- |
| 23 | Tension in Cable | 49 |
| 24 | Tow Mechanism Schematic | 50 |
| 25 | Orbiter Hook Mechanism | 52 |
| 26 | Orbiter Subsonic Trimmed Lift Coefficient | 58 |
| 27 | Orbiter Subsonic Trimmed Drag Coefficient | 59 |
| 28 | Orbiter Subsonic Trimmed Lift-to-Drag Ratio | 60 |
| 29 | Orbiter Hypersonic Trimmed Lift Coefficient | 61 |
| 30 | Orbiter Hypersonic Trimmed Drag Coefficient | 62 |
| 31 | Orbiter Hypersonic Trimmed Lift-to-Drag Ratio | 63 |
| 32 | Orbiter Maneuver Capability | 72 |
| 33 | Typical Orbiter-Booster Rendezvous | 73 |
| 34 | Orbiter and Booster Reentry | 74 |
| 35 | Airplane Maneuver Capability | 76 |
| 36 | Axis System | 77 |
| 37 | Airplane Motion Relative to Orbiter (500 to 200 Seconds-to-Go) | 78 |
| 38 | Airplane Motion Relative to Orbiter (Final 200 Seconds) | 79 |
| 39 | Orbiter Subsonic Glide | 80 |
| 40 | Orbiter-Booster Relative Motion (Range) | 82 |
| 41 | Orbiter-Booster Relative Motion (Altitude) | 83 |
| 42 | Orbiter Motion Relative to Airplane (Five to Two Minutes-to-Go) | 87 |
| 43 | Control of Relative Altitude | 88 |

## LIST OF TABLES


| Table No. | | Page No. |
|---|---|---|
| 1 | Orbiter Weight Data | 17 |
| 2 | Booster Weight Data | 22 |
| 3 | Mission Weight Data | 24 |
| 4 | Other Concepts Considered | 26 |
| 5 | Tow Concept Weights | 53 |
| 6 | Rendezvous Phases | 86 |
| 7 | Recommended Future Studies | 94 |
| A-1 | Orbiter Recovery Concepts | 96 |

## SYMBOLS AND ABBREVIATIONS

| | |
|---|---|
| a | length of major axis of orbiter body cross-section |
| A/C | aircraft |
| ACS | attitude control system |
| AR | aspect ratio |
| b | length of minor axis of orbiter body cross section |
| cap. | capacity |
| $C_D$ | coefficient of drag |
| c.g. | center-of-gravity |
| conver. | conversion |
| $C_L$ | coefficient of lift |
| ℄ | centerline |
| $C_{L\alpha}$ | lift curve slope |
| D | drag, lb. |
| deg | degrees |
| dist. | distribution |
| DSI | Dallas Support Item |
| EAS | equivalent airspeed |
| $E_D$ | energy absorbed in docking, lb. ft. |
| elect. | electrical |
| environ. | environmental |
| F | force, lb |
| ft. | feet |
| gal. | gallons |
| gpm | gallons per minute |

SYMBOLS AND ABBREVIATIONS (Continued)

| | |
|---|---|
| GLOW | gross lift-off weight |
| hydra. | hydraulic |
| in. | inches |
| INCOMPRESS | incompressible |
| L | Lift, lb. |
| l | length |
| lb | pounds |
| lbs | pounds |
| L/D | lift-to-drag ratio |
| $LH_2$ | liquid hydrogen |
| LOX | liquid oxygen |
| LTV | Ling-Temco-Vought |
| M | Mach Number |
| m | mass, slugs |
| maneuv. | maneuver |
| max. | maximum |
| min. | minutes |
| MN | Mach Number |
| NASA | National Aeronautics and Space Administration |
| n.mi. | nautical miles |
| O.D. | outside diameter |
| OMS | orbital maneuver system |
| psf | pounds per square foot |
| psi | pounds per square inch |
| q | dynamic pressure, $lb/ft^2$ |

## SYMBOLS AND ABBREVIATIONS (Continued)

| | |
|---|---|
| R | range, nautical miles |
| Ref. | reference |
| $S_{REF}$ | reference area, $ft^2$ |
| $S_{TAILS}$ | area of horizontal tails, $ft^2$ |
| sec | seconds |
| SR | specific range, n.mi./lb |
| t | cable tension, lb; time, seconds |
| TN | NASA Technical Note |
| V | velocity, ft/sec; airspeed, knots |
| VMSC | Vought Missiles and Space Company |
| W | weight, lb |
| wt | weight |
| $\dot{W}$ | fuel flow rate, lb/hr |
| $W_F$ | weight of fuel consumed, lb. |
| Wt | weight |
| X | body station, fraction of length |
| X | |
| Y | axis system coordinates |
| Z | |
| Z | relative altitude |
| Z | rate of change of relative altitude, ft/sec |
| Zr | zirconium |
| $ZrO_2$ | zirconium oxide |
| Zr REI | zirconium reentry insulation |
| $\alpha$ | angle-of-attack, degrees |

## SYMBOLS AND ABBREVIATIONS (Continued)

| | |
|---|---|
| $\Delta$ | incremental change |
| $\gamma$ | flight path angle, degrees |
| $\pi$ | 3.1416 |
| $\theta$ | angle between tow cable and horizontal, degrees |
| & | and |
| @ | at |
| ° | degrees |
| °F | degrees Fahrenheit |
| ' | feet |
| " | inches |
| $\int$ | integral |
| $\int Fdt$ | impulse, lb. sec. |
| % | percent |
| / | per |
| x | by |
| = | is equal to |
| $\simeq$ | is approximately equal to |
| $\cong$ | is approximately equal to |
| < | is less than |
| > | is greater than |

ATMOSPHERIC RENDEZVOUS
FEASIBILITY STUDY

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## SUMMARY

A study was carried out to determine the feasibility of using atmospheric rendezvous to increase the efficiency of space transportation and to determine the most effective implementation. It is concluded that atmospheric rendezvous is feasible and can be utilized in a space transportation system to reduce size of the orbiter vehicle, provide a powered landing with go-around capability for every mission, and achieve lateral range performance that exceeds requirements. A significantly lighter booster and reduced launch fuel requirements are additional benefits that can be realized with a system that includes a large subsonic airplane for recovery of the orbiter. Additional reduction in booster size is possible if the airplane is designed for recovery of the booster by towing. An airplane about the size of the C-5A is required.

NASA Space Shuttle data were used to define baseline configurations and weights for this study. Weight of the orbiter at rendezvous is about 200,000 pounds.

Two basic approaches were investigated for performing the rendezvous and recovery tasks. One approach considers use of a large airplane with which rendezvous occurs after the orbiter has completed its hypersonic glide and has slowed to subsonic flight conditions. The other approach involves use of a recoverable booster which may rendezvous with the orbiter at any speed up to its maximum burnout speed. The booster may launch an orbiter and recover another orbiter on the same flight. Although feasible, the orbiter-booster rendezvous is less attractive than the orbiter-airplane case. Booster cruise and landing with a docked orbiter aboard requires increased booster wing area and cruise propulsion, resulting in a larger booster than that defined by the Phase B Space Shuttle studies. Other disadvantages of orbiter-booster rendezvous are:

(1) The booster has a launch window of only one minute for this type of mission.

(2) Booster propulsion or orbiter drag control is required during the docking operation.

(3) The booster cruise range must be increased (compared to that for Phase B studies) or it must be permitted to land downrange from the launch site.

(4) Design of a docking system is more difficult for the hypersonic case because of the more severe thermal environment.

Two conceptual designs are defined for recovery of the orbiter by the airplane. One of these involves docking the vehicles, lower surface of the orbiter to top of the airplane. The primary docking component is a large telescoping boom on the airplane. The boom supports a latching mechanism, which locks into a docking cone on the orbiter. The technique is similar to an air-to-air refueling operation.

The other conceptual design involves towing of the orbiter by the airplane. Operation is somewhat similar to air-snatch of a parachuting payload. The orbiter is towed to the landing site, and then lands in the towed condition. This results in a lower landing speed than for an unpowered landing, permits landing at very small sink rates, and provides go-around capability.

Additional airplane weight required for either of these recovery concepts is well within the cargo capability of the airplane.

Rendezvous guidance requirements can be satisfied with inertial and radar guidance components, on-board computers, and a communication system. Visual observations as well as radar data are used during approach to the position required to initiate docking or towing. Relative motion of the two vehicles during the final two minutes of rendezvous is very similar to the Apollo Lunar Module's approach to the Lunar surface.

Orbiter-airplane docking is initiated at an altitude of about 32,000 feet. A time increment of approximately three and one-half minutes is available to complete the docking phase, which is satisfactory based on design of the docking system and comparison with air-to-air refueling experience.

Further development of the atmospheric rendezvous concept requires studies to provide preliminary design data for vehicles and docking or towing systems, analysis of the dynamics of docking or towing operations, development of a rendezvous guidance technique, manned simulation studies of rendezvous and docking or towed landing, crew safety and abort studies, and cost effectiveness studies.

## 1.0 INTRODUCTION

The national goal of developing low cost space transportation for near earth support of orbiting research laboratories and assembly areas for deeper space excursions has focused attention on reusable vehicles and efficiency of system operation. Several reusable launch and entry vehicle designs have been extensively studied, and it is readily apparent that reductions in structural mass fractions, and large lateral range capabilities are important factors in realizing an efficient space logistics system.

Studies at the NASA Langley Research Center determined that one approach to achieving increased payload mass fraction and more efficient utilization of boosters is an atmospheric rendezvous concept for recovery of payload vehicles. This concept is outlined in Reference 1, which also includes a summary of some of the results from this study. The large weight increment associated with providing a reentry payload vehicle with the capability to make a conventional landing is reduced by replacing such equipment as wings, engines, fuel, and landing gear with equipment required for acquisition by a carrier vehicle. The carrier could be an airplane or a recoverable booster stage which flies the payload vehicle to an appropriate landing site. Recovery by an airplane would probably be at subsonic speed. Recovery by a booster stage could be at any speed up to its normal burnout speed. Programs requiring many launches and recoveries could schedule operations such that a recoverable booster stage on a single flight provides first stage boost for a new payload, then recovers a returning payload by rendezvous and docking in the atmosphere. Good mission flexibility is attained, in terms of recovery range and choice of landing site, in recovery by either aircraft or booster.

The Langley studies provided basic groundrules and constraints for the feasibility study reported here. It was specified that both of the above types of rendezvous would be investigated. Subsonic rendezvous and recovery by an airplane are illustrated in Figure 1. Orbiter retrieval by a recoverable booster is shown in Figure 2. In both cases the carrier vehicle acquires the orbiter vehicle and transports it to the landing site. Acquisition methods include docking and towing.

### 1.1 Historical Background

Few of the atmospheric rendezvous, retrieval, and carrying techniques discussed in this report are really new. In 1929 the capability of dirigibles to carry fighter planes was being developed (Figure 3). The system was operational in the nineteen-thirties. Figure 4 (from Reference 2) shows a Curtis F9C docked to the U.S.S. Macon. Many successful hook-ups and releases were made. Additional details may be found in Reference 2.

A very long range reconnaissance capability was developed in the fifties by using modified B-36 and F-84 aircraft. A photograph of that system is shown in Figure 5. (See Reference 3 for additional details.)

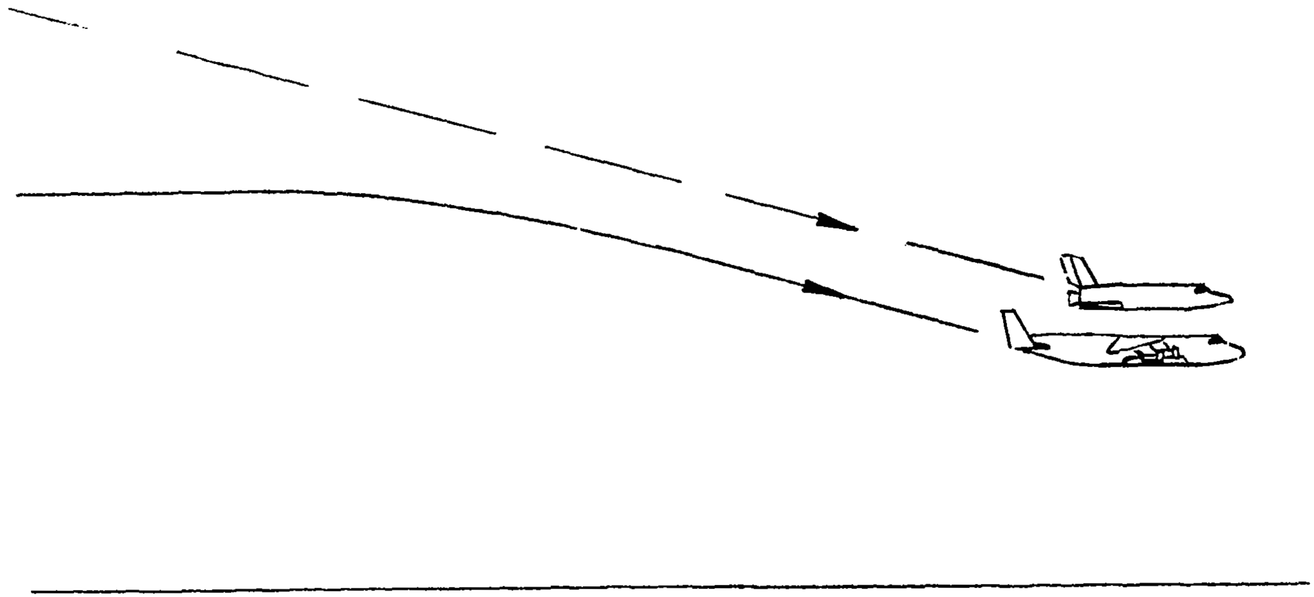


Figure 1. - Orbiter-Airplane Rendezvous

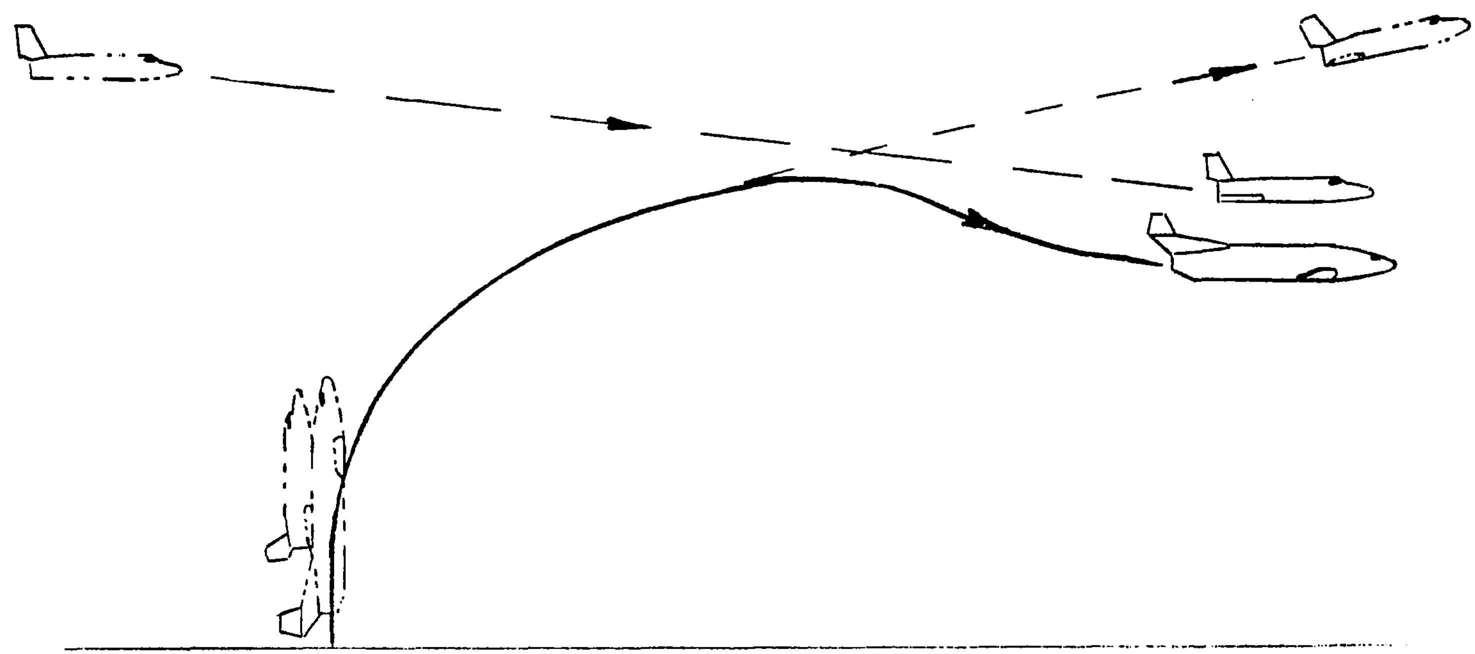


Figure 2. - Orbiter-Booster Rendezvous

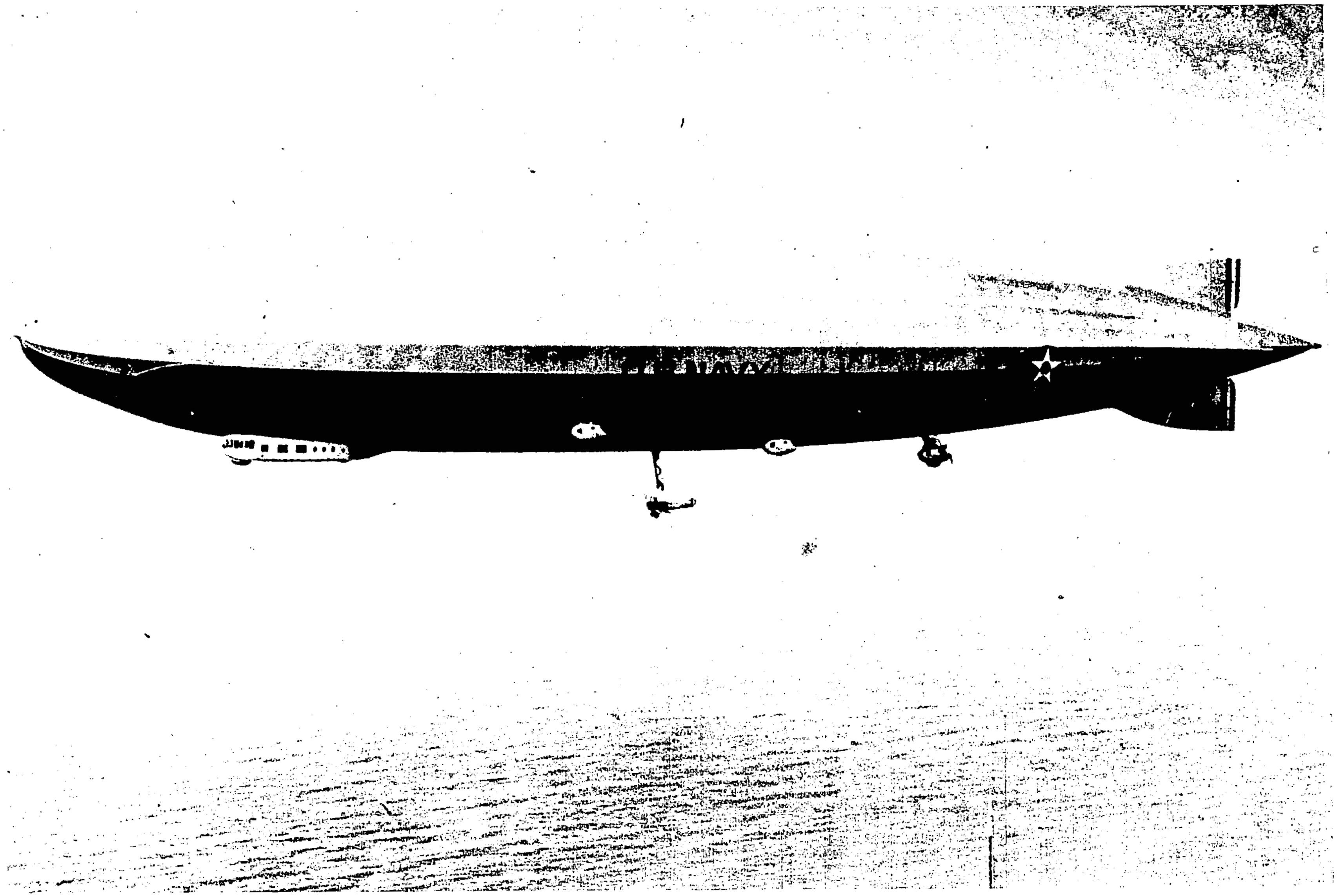


Figure 3. - Vought UO-1, U.S.S. Los Angeles (August, 1929)

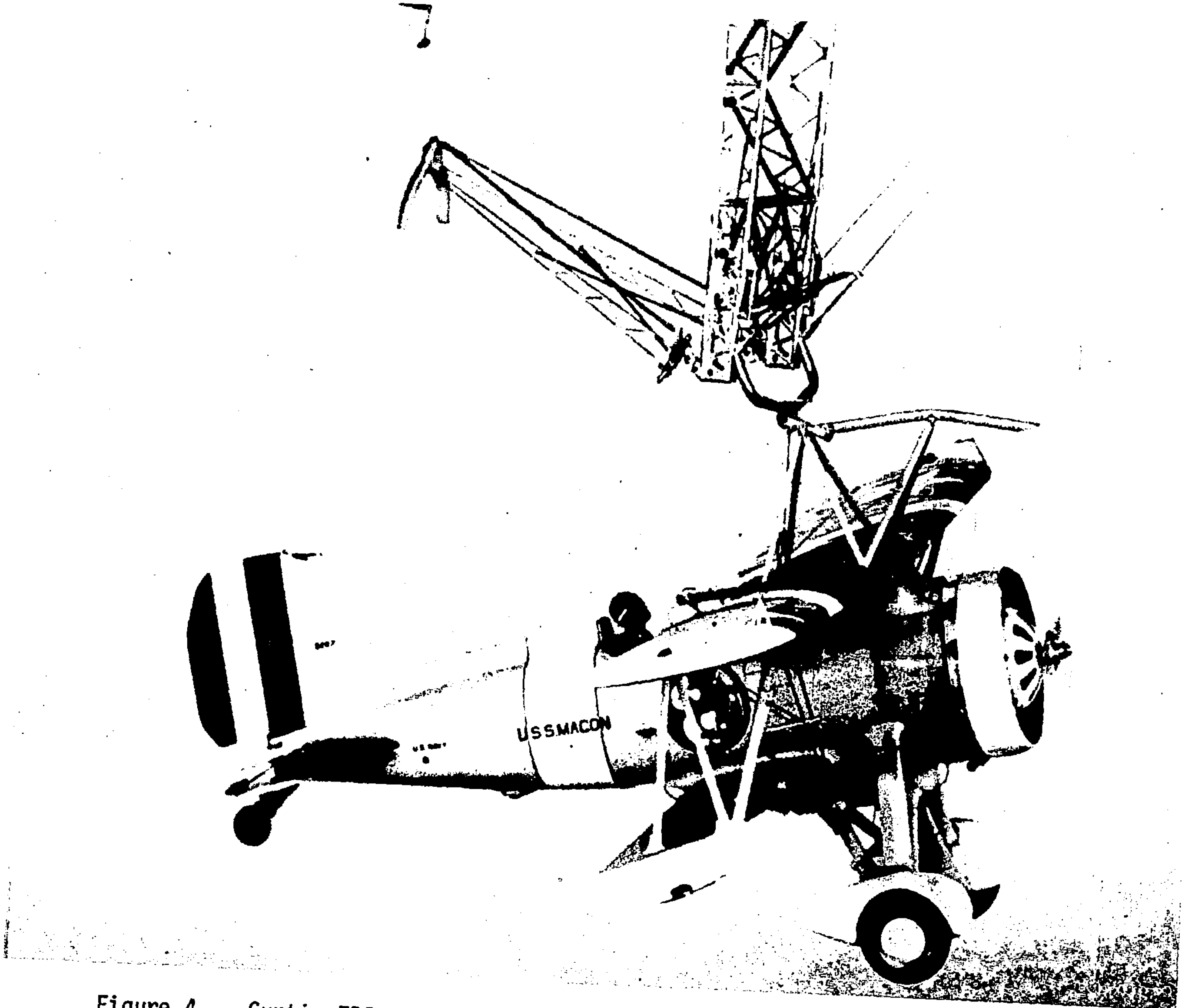



Figure 4. - Curtis F9C, U.S.S. Macon (1933) (Courtesy Air Force Museum, Wright-Patterson Air Force Base, R. L. Cavanagh Photo Collection)

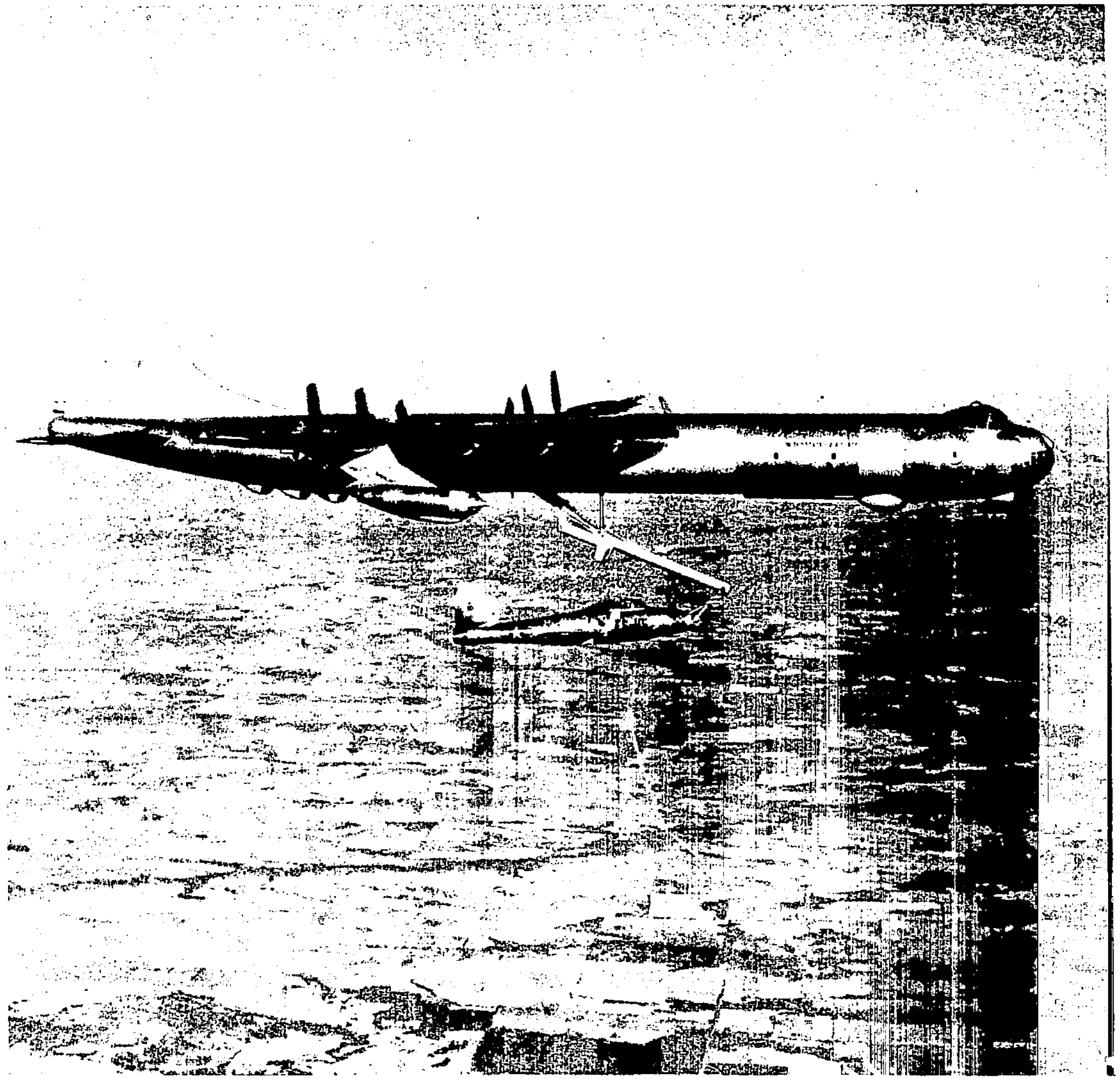


Figure 5. - RB-36, (YR)F-84F (1956) (Courtesy Air Force Museum, Wright-Patterson Air Force Base)

Airplanes of the type shown in Figure 6 have been used in recent years to snatch and retrieve small reentry payloads during parachute descent in the atmosphere.

Two long-range flights were made in 1938 by a dual seaplane configuration, the Short-Mayo Composite (Reference 4). A relatively small seaplane, the Mercury, was carried through takeoff and initial climb by a large flying-boat, the Maia. The Mercury separated from the upper fuselage of the Maia at cruise altitude. One flight, from Dundee to South Africa, established an international distance record for seaplanes. No rendezvous and docking were involved; however, the carrying technique is similar to that considered in this study.

## 1.2 Objectives

The purpose of this study is to determine the feasibility of using atmospheric rendezvous to increase efficiency of orbital operations and space transportation and to determine the most effective implementation of the atmospheric rendezvous mode.

Other objectives are to:

(1) Define the benefits of atmospheric rendezvous.

(2) Provide conceptual designs of the payload vehicle and docking system.

(3) Compute trajectories and relative velocities.

(4) Define rendezvous and docking techniques.

(5) Estimate contact velocities and forces during docking.

(6) Determine the flexibility of landing sites provided by this mode of operation.

(7) Define sensor, guidance, and control requirements.

The study statement of work specifies the following tasks:

(1) Conceptual design will be performed to provide configuration data for the study. Optimization of configurations is not required; however, design studies will be of sufficient depth to provide realistic baseline configurations for the payload vehicle and docking equipment. Weight estimates will be made for the payload vehicles. This task will also provide definitions of baseline aircraft and booster carrier vehicles. Results of preliminary studies by NASA and other data provided by NASA will assist in this task. In defining the reentry vehicle

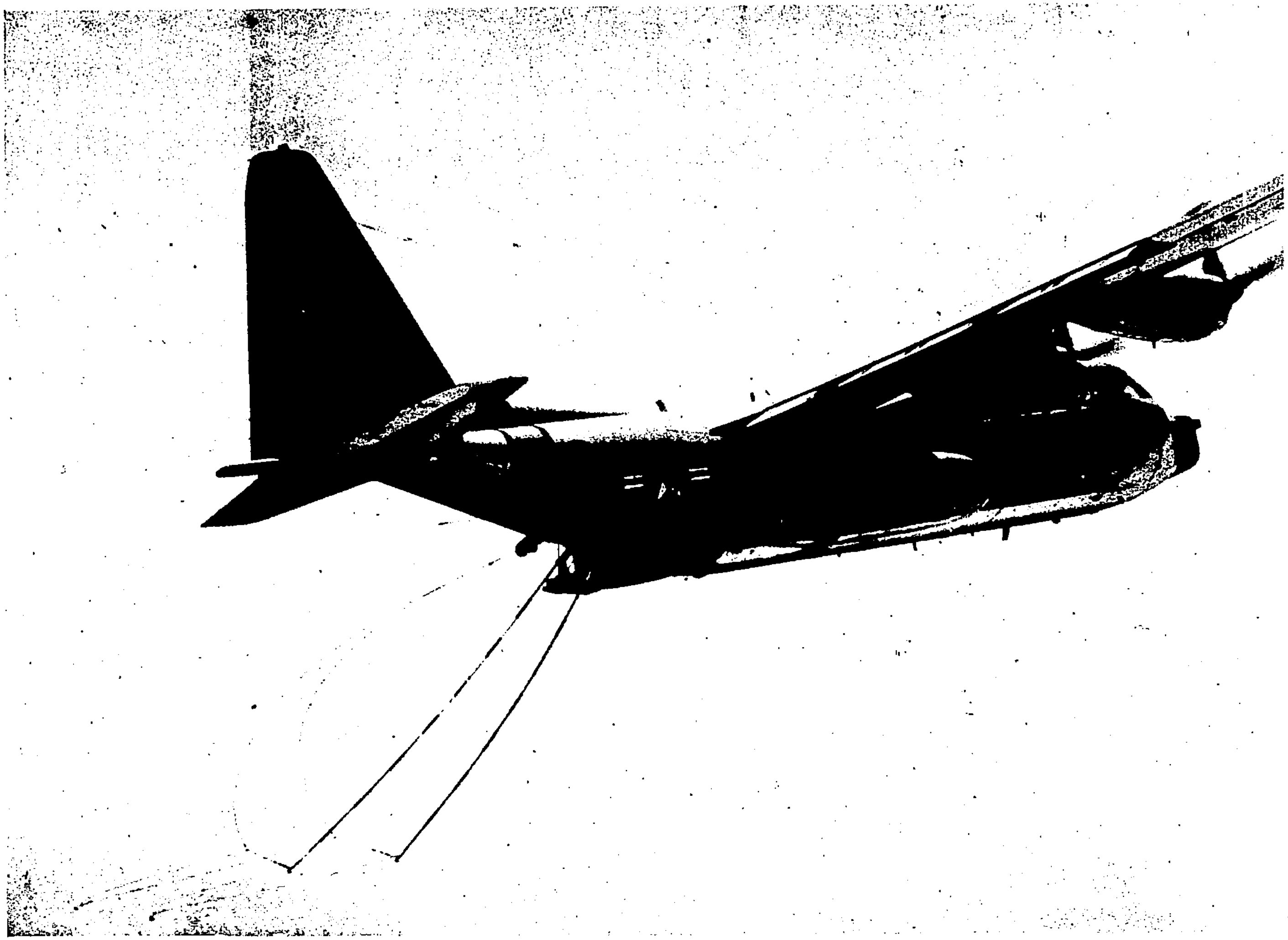


Figure 6. - HC-130 Recovery of Atmospheric Sampling Capsules
(Photograph published by All American Engineering Company)

configuration, a major objective is to maximize lift-to-drag ratio (up to the point of incurring a significant weight penalty) so that rendezvous and docking can be performed at small flight-path angles and large available time increments. Approximately equal effort will initially be placed on study of rendezvous with aircraft and booster stages; however, once a superior mode of operation is identified, major emphasis will be on that mode.

(2) Trajectories will be computed for boost and reentry vehicles, and aircraft flight characteristics will be defined. Relative velocities will be determined for rendezvous and docking at various points along the reentry-glide trajectory. Performance in terms of lateral range capability after docking will be determined for the carrier vehicles.

(3) Docking techniques will be postulated and studied to determine satisfactory approaches. The major effort will be to find methods and designs that can cope with aerodynamic loads at all speeds considered and heating problems at hypersonic speeds. Stability of the coupled vehicles will also be considered. Contact velocities and forces will be determined based on trajectory data and docking technique.

(4) General requirements of all subsystems will be defined consistent with selected rendezvous and docking techniques. Included are types of sensors, guidance and control logic, control forces, and impulse requirements.

(5) Advantages, disadvantages, problem areas, and approaches to solutions will be identified for atmospheric rendezvous based on results of the above tasks.

## 1.3 Major Assumptions

Booster and orbiter configuration and weight data used in this study are based on NASA Space Shuttle studies. Preliminary Phase B results from both McDonnell-Douglas and North American studies are used, as well as some data from various Phase A studies. Another possible application for atmospheric rendezvous is for recovery of a very high altitude hypersonic cruise vehicle. It is anticipated that other future programs will provide additional possibilities. The Space Shuttle program is utilized to a great extent in this study because it is the only current program that can provide extensive baseline data.

Payload weight requirements and size of the orbiter cargo bay are the same as for Phase B Space Shuttle studies. Orbiter size and weight are minimized by initially providing no wings, landing gear, cruise engines, or fuel. The orbiter body is shaped to maximize lift-to-drag ratio, within the constraint of maintaining a near-minimum structural weight ratio. Minimum weight tail surfaces are provided for stability and control. The orbiter

configuration also contains equipment required for docking or towing by the carrier vehicle.

The recovery airplane is assumed to be similar to the C-5A, because it is obviously an advantage to utilize an aircraft as large and powerful as possible due to the large size of the vehicle to be recovered.

Rendezvous at supersonic and hypersonic speeds is emphasized for the orbiter-booster rendezvous case. It is assumed that subsonic recovery of the orbiter could be better achieved by an airplane because of superior subsonic cruise performance compared to that for the booster.

## 2.0 BASELINE CONFIGURATIONS

Investigation of the feasibility of atmospheric rendezvous requires defining reasonable configurations for the vehicles involved. The paragraphs which follow describe these baseline vehicles and the rationale followed in establishing the configurations used.

### 2.1 Orbiter

Initially, it was assumed that the orbiter would not be required to land independently, and therefore the wing, landing gear and cruise propulsion system would be unnecessary. The remaining systems identified in the Phase A & B shuttle studies would be required, however, as well as the orbiter's ascent propulsion system and payload capacity. In order to obtain good equilibrium glide characteristics the body was reshaped to obtain a reasonably high lift-to-drag ratio (L/D). The rendezvous operation between the orbiter and its carrier vehicle requires a controllable, stable vehicle. Minimum size stabilizing and control surfaces were added to satisfy this requirement.

From data available in-house at Vought Missiles and Space Company relative to the Phase B shuttle studies and data provided by NASA Langley from the Phase A studies, the volume of propellants, engines, equipment and crew as well as the payload was estimated. Using these volumes and considering the aerodynamic requirements, a minimum size vehicle was configured.

The general arrangement of the resulting baseline vehicle concept is shown in Figure 7, with its pertinent dimensions and area data. The aerodynamic and performance characteristics of the vehicle along with its application to selected rendezvous and retrieval concepts are discussed in later sections. The docking cones shown in Figure 7 are applicable to the hard dock concept only. In the case of the towing concept a retractable hook is used and installed above the crew compartment. These details are discussed in Section 4.2 of this report.

Orbiter weight was initially estimated based on preliminary data from Phase B studies by both McDonnell-Douglas and North American and Phase A studies by North American. Subsystem weights from these sources were reviewed and compared, and values were selected or scaled based on judgement. The resulting estimated orbiter weight at rendezvous was approximately 180,000 pounds. It was then decided to prepare another estimate based on inputs from only one source to provide a better comparison with a specific Phase B configuration. The McDonnell-Douglas data were used because these data were available to this study in somewhat greater detail. The resulting estimated rendezvous weight for the orbiter was 210,000 pounds. A comparison of orbiter weight data is shown in Table 1, indicating a weight at rendezvous approximately 50,000 pounds less than the landing weight for a Phase B orbiter.

| AREA DATA | |
|---|---|
| HORIZONTAL STABILIZER (TOTAL EXPOSED) | 1000 $FT^2$ |
| VERTICAL STABILIZER (EXPOSED) | 610 $FT^2$ |
| BODY PROJECTED PLANFORM | 6350 $FT^2$ |
| TOTAL PLANFORM | 7350 $FT^2$ |

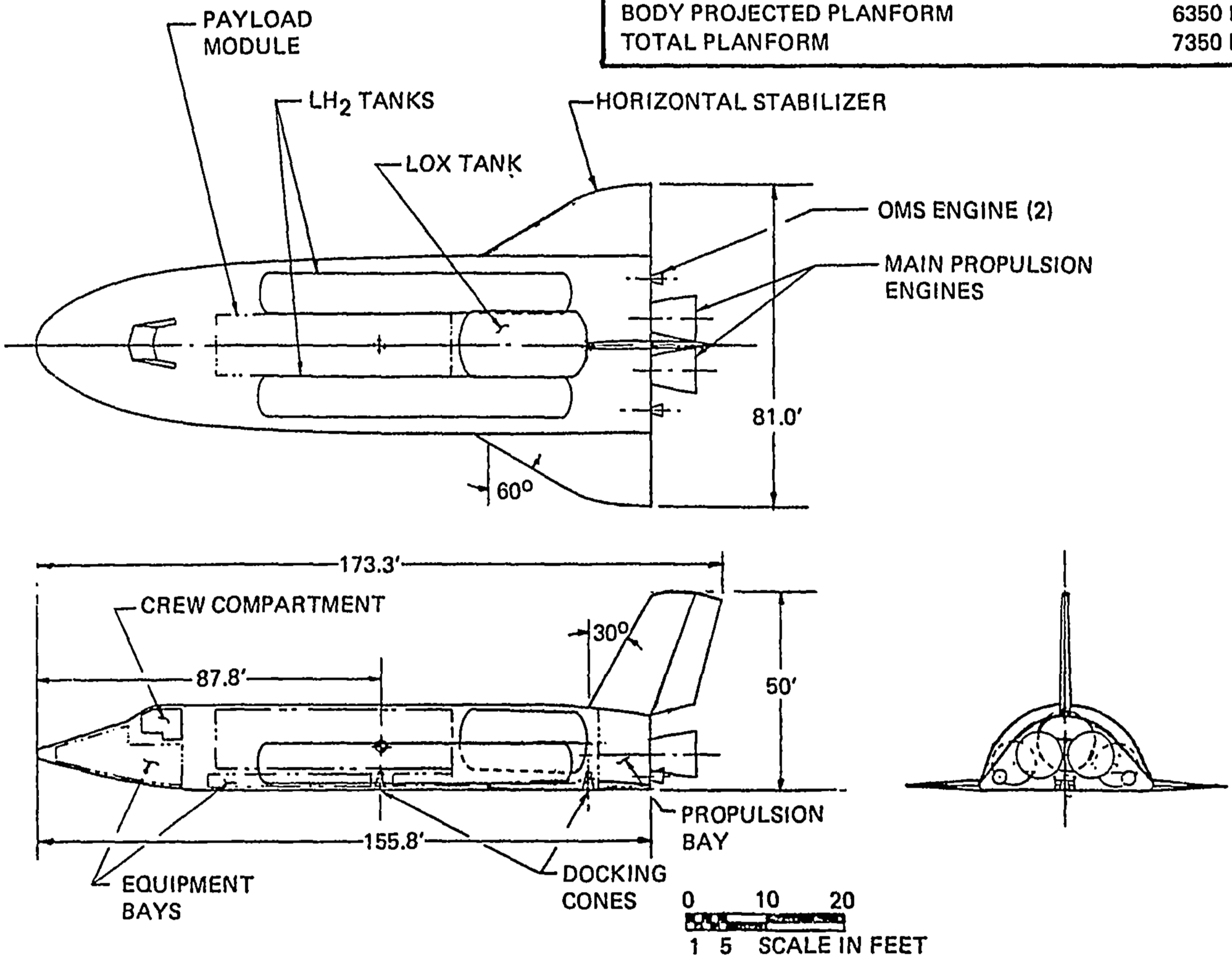



Figure 7. - General Arrangement, Orbiter Concept

TABLE 1

ORBITER WEIGHT DATA

| Configuration | McDonnell-Douglas Phase B | Atmospheric Rendezvous |
|---|---|---|
| Weights, lb | | |
| Wing group | 28,311 | 0 |
| Tail Group | 5,790 | 12,000 |
| Body Group | 62,421 | 55,000 |
| Induced Envir. Protection | 32,496 | 25,000 |
| Landing, recovery, docking | 8,963 | 1,500 |
| Propulsion, ascent | 25,275 | 25,300 |
| Propulsion, cruise | 400 | 0 |
| Propulsion, auxiliary | 11,805 | 7,000 |
| Prime Power | 1,466 | 1,500 |
| Elect. Conver. and Dist. | 1,364 | 1,400 |
| Hydra. Conver. and Dist. | 1,782 | 1,800 |
| Surface Controls | 2,981 | 2,000 |
| Avionics | 4,365 | 4,400 |
| Environmental Control | 7,088 | 7,100 |
| Personnel Provisions | 210 | 200 |
| Growth/Uncertainty | 17,641 | 13,000 |
| Subtotal (dry weight) | (212,358) | (157,200) |
| Personnel | 400 | 400 |
| Cargo | 79,653* | 79,700* |
| Residual fluids | 3,786 | 3,000 |
| Subtotal (inert weight) | (296,197) | (240,300) |
| Reserve Fluids | 16,482 | 14,000 |
| Inflight losses | 10,294 | 8,400 |
| Propellant-ascent | 523,794 | 426,000 |
| Propellant-cruise | 0 | 0 |
| Propellant-Maneuv/ACS | 12,616 | 10,200 |
| Total (ignition weight) | (859,383) | (698,900) |
| Injection | 334,117 | 272,000 |
| Rendezvous or landing | 266,488* | 210,000* |

*Landed cargo is 40,000 lb.

Initial weight is reduced by approximately 160,000 pounds. Ascent propellant is reduced by almost 100,000 pounds.

Recent studies of shuttle vehicles utilizing expendible boosters and orbiter drop tanks result in large variation of orbiter landing weight, from as low as 100,000 pounds to values somewhat greater than those for Phase B. Most of the results of this study are based on a rendezvous weight of 200,000 pounds. It is felt that this is representative, and possibly somewhat conservative considering current efforts to reduce system weight. A 20% reduction in recovery weight due to the atmospheric rendezvous concept, as indicated by Table 1, is probably also representative.

## 2.2 Recovery Airplane

As previously stated the Lockheed C-5A, "Galaxy" was selected as the baseline airplane configuration for the study, primarily because of its size and payload capacity of 265,000 pounds. The aircraft has a design landing weight of 635,000 pounds and a maximum landing weight of 769,000 pounds. The dimensions of the main cargo compartment are: length 121 feet, height 13.5 feet, and width 19.0 feet. A general arrangement with its principal external dimensions is included as Figure 8.

## 2.3 Boosters

The McDonnell-Douglas Phase B booster was chosen as the baseline for the study because more data were available at VMSC on this vehicle, and its twin vertical tail configuration might provide some advantages for orbiter-booster docking. Figure 9 is a general arrangement of this vehicle.

This booster was examined to determine the effects of launching a lighter orbiter and landing with the orbiter docked to it. This resulted in an increased wing area to hold the wing loading to its original value on landing and an increase in cruise propulsion capability to handle the additional drag imposed by the orbiter during cruise back to a landing site. The control surfaces, canard and verticals were also resized to retain essentially the same tail volumes as the original configuration. The configuration which resulted along with its major dimensions is shown in Figure 10.

Estimated weights for these boosters and comparisons with Phase B data are shown in Table 2. Several types of atmospheric rendezvous boosters are considered.

(1) The booster capable of a conventional landing and sized to launch an orbiter that is recovered by an airplane is significantly lighter than a comparable Phase B booster, and uses approximately 450,000 pounds less fuel for launch.

(2) The booster designed to recover the orbiter is larger than the Phase B booster due to increased wing area and cruise propulsion capability. The combination of smaller orbiter and larger

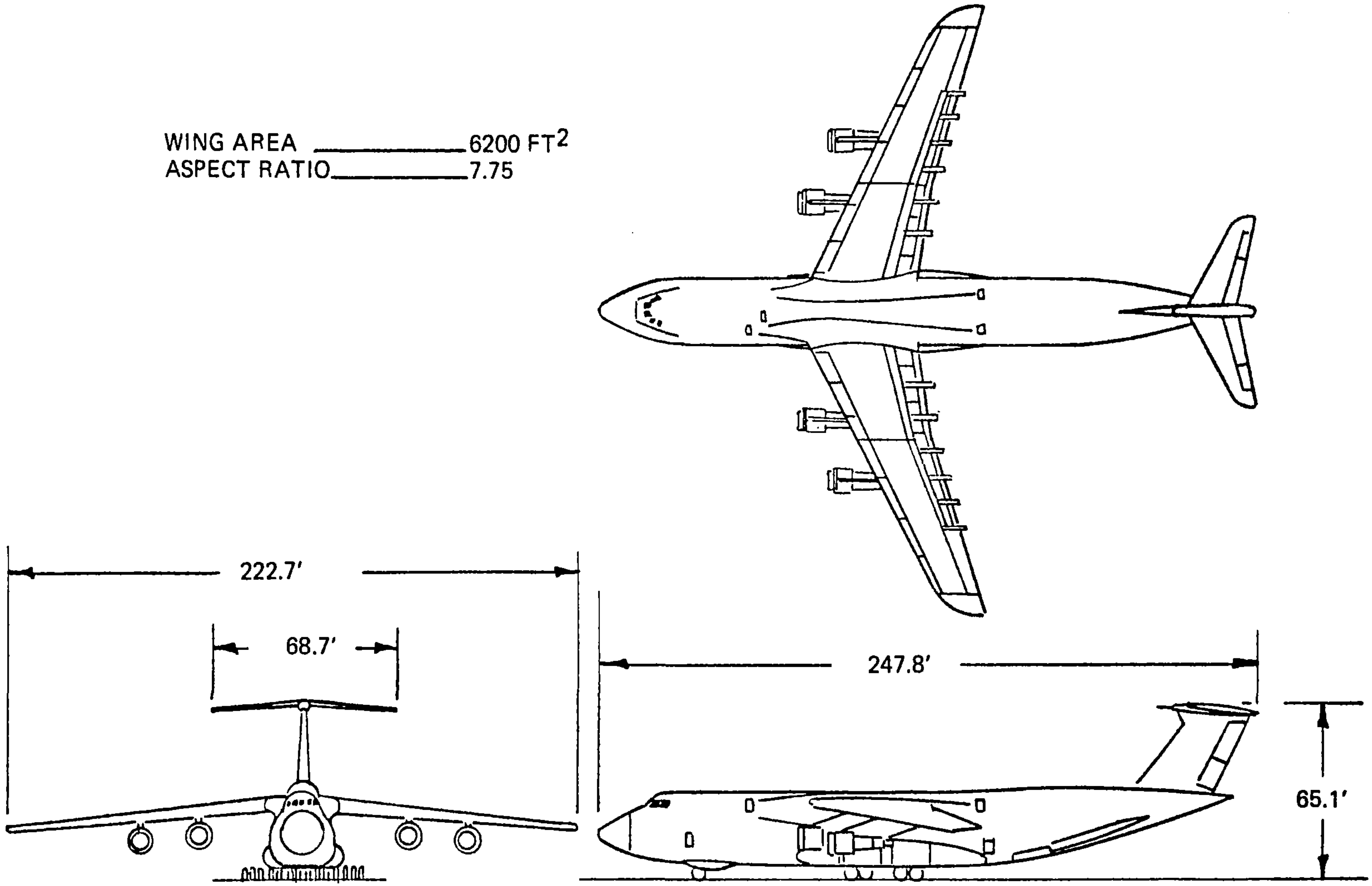



Figure 8. - General Arrangement, Lockheed C-5A

| GEOMETRIC DATA | |
|---|---|
| WING AREA (TOTAL) | 6016 $FT^2$ |
| CANARD AREA (TOTAL) | 1660 $FT^2$ |
| VERTICAL TAIL (EACH) | 438 $FT^2$ |

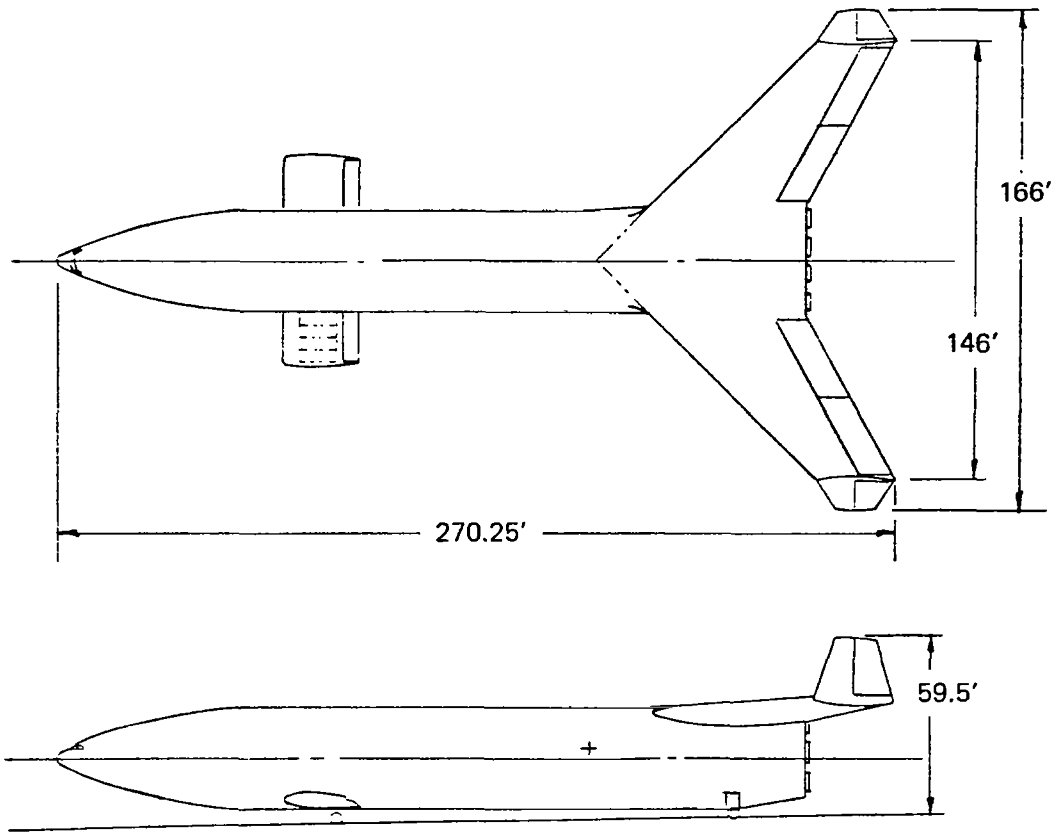



Figure 9. - General Arrangement, McDonnell-Douglas Phase B Booster

| GEOMETRIC DATA | |
|---|---|
| WING AREA (TOTAL) | 9200 $FT^2$ |
| CANARD AREA (TOTAL) | 2270 $FT^2$ |
| VERTICAL TAIL (EACH) | 643 $FT^2$ |

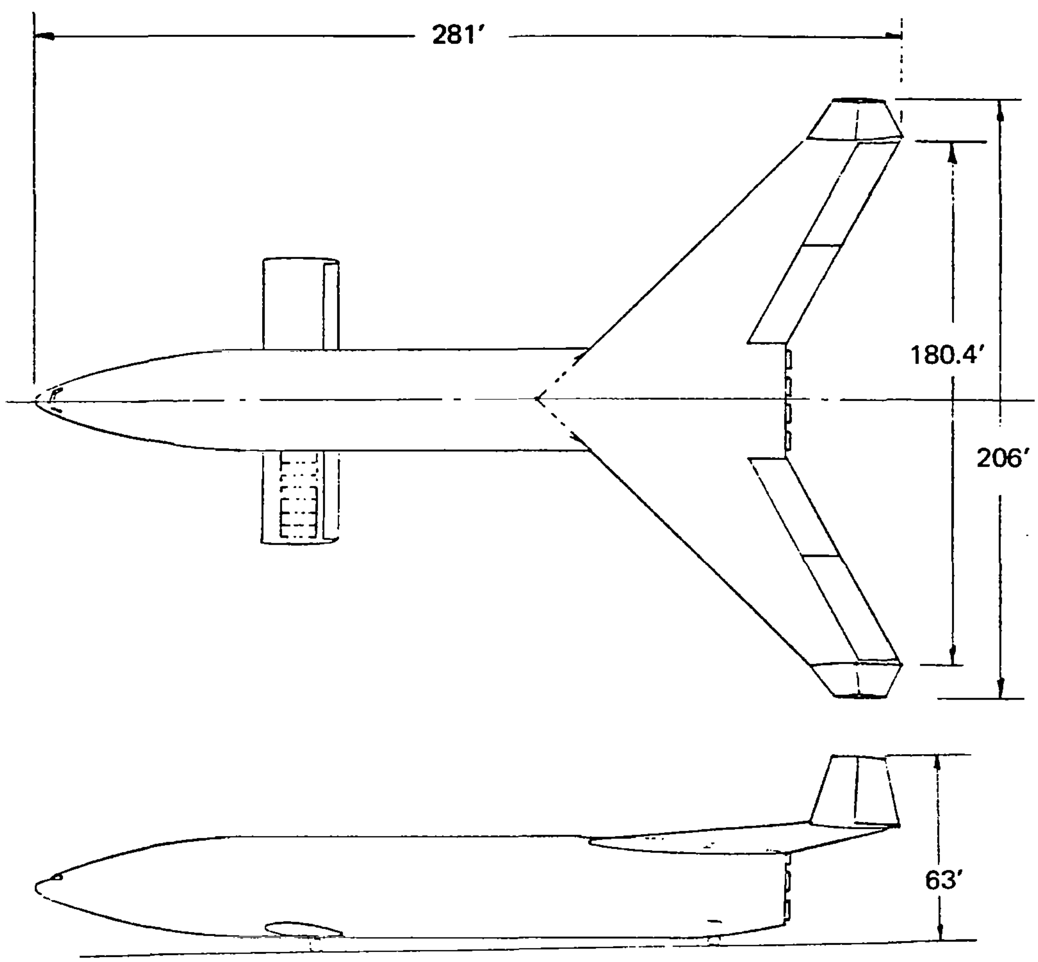



Figure 10. - General Arrangement, Modified Booster

TABLE 2

BOOSTER WEIGHT DATA

| CONCEPT | McDONNELL-DOUGLAS PHASE B | ATMOSPHERIC RENDEZVOUS | | |
|---|---|---|---|---|
| Orbiter Recovery | LAND | AIRPLANE | BOOSTER | AIRPLANE |
| Booster Recovery | LAND | LAND | LAND | AIRPLANE TOW |
| Weights, lb. | | | | |
| Wing Group | 51,458 | 46,300 | 76,000 | 46,300 |
| Tail Group | 14,839 | 13,400 | 22,000 | 13,400 |
| Body Group | 151,543 | 128,800 | 140,000 | 128,800 |
| Induced Environ. Protection | 66,510 | 56,500 | 70,000 | 56,500 |
| Landing, Recovery, Docking | 23,142 | 20,900 | 34,000 | 20,900 |
| Propulsion-Ascent | 117,101 | 117,100 | 117,100 | 117,100 |
| Propulsion-Cruise | 36,925 | 36,900 | 55,000 | 0 |
| Surface Controls | 4,910 | 4,400 | 7,300 | 4,400 |
| Other dry weight | 53,981 | 47,800 | 58,700 | 47,800 |
| Subtotal (dry weight) | (520,409) | (472,100) | (580,100) | (435,200) |
| Personnel | 400 | 400 | 400 | 400 |
| Residual fluids | 9,760 | 8,300 | 9,000 | 8,300 |
| Subtotal (inert weight) | (530,569) | (480,800) | (589,500) | (443,900) |
| Propellant-Ascent | 3,064,000 | 2,615,000 | 2,970,000 | 2,345,000 |
| Propellant-Cruise | 110,000 | 100,000 | 165,000 | 0 |
| Other fluids and losses | 74,591 | 63,400 | 71,200 | 63,400 |
| Total (ignition weight) | (3,779,160) | (3,259,200) | (3,795,700) | (2,852,300) |
| Burnout | 715,160 | 644,200 | 825,700 | 507,300 |
| Start Cruise | 692,187 | 625,000 | 804,000 | --- |
| Rendezvous or Landing | 542,100 | 495,000 | 601,000 | 457,000 |

booster results in a decrease in booster launch fuel required of approximately 100,000 pounds compared to Phase B.

(3) A launch system for which both orbiter and booster are recovered by aircraft is shown to be the smallest of the combinations studied. Booster launch fuel is about 700,000 pounds less than Phase B for this case.

Mission weights for various combinations of orbiters and boosters are shown in Table 3. Significant reductions in gross lift-off weight (compared to Phase B) are shown for concepts that include orbiter-airplane rendezvous. Reduction in gross lift-off weight is small for orbiter-booster rendezvous; however, this approach could be attractive for a space transportation system that involves many orbiters and relatively few boosters, due to reduced weight of the orbiter.

Credibility of Weight Estimates

Since the primary purpose of this study was to investigate feasibility of atmospheric rendezvous, a relatively small effort was allocated for vehicle weight estimates. Estimated orbiter and booster weights presented in this report are based on preliminary subsystem incremental weight data from Phase B studies. Simple and generally conservative methods were used in adjusting these increments to the smaller vehicles required for atmospheric rendezvous. For example, ascent propulsion components were assumed to be sized by Phase B studies, and were not scaled down (see Tables 1 and 2). In the preparation of Reference 1, which summarizes results from this study and related work at the NASA Langley Research Center, some refinement of these estimates was achieved, such as allowing a decrease in ascent propulsion weight as overall vehicle size decreases. As a result, vehicle weights shown in Reference 1 are generally five to fifteen percent less (and in the case of rendezvous weight of a booster towed by an airplane, twenty-five percent less) than those presented in this report. However, these differences do not affect general weight trends and conclusions based on this study. More accurate quantitative evaluation of benefits of atmospheric rendezvous from a vehicle weight standpoint would require preliminary design studies and more detailed weight analyses.

TABLE 3

MISSION WEIGHT DATA

| CONCEPT | McDONNELL-DOUGLAS PHASE B | ATMOSPHERIC RENDEZVOUS | | |
|---|---|---|---|---|
| ORBITER RECOVERY | LAND | AIRPLANE | BOOSTER | AIRPLANE |
| BOOSTER RECOVERY | LAND | LAND | LAND | AIRPLANE TOW |
| Weights, lb. | | | | |
| Liftoff | | | | |
| Booster | 3,779,160 | 3,259,200 | 3,795,700 | 2,852,300 |
| Orbiter | 859,383 | 698,900 | 698,900 | 698,900 |
| Total (GLOW) | 4,638,543 | 3,958,100 | 4,494,600 | 3,551,200 |
| Orbiter Rendezvous or Landing | 266,488 | 210,000 | 210,000 | 210,000 |
| Booster Rendezvous or Landing | | | | |
| Booster Alone | 542,100 | 495,000 | 601,000 | 457,000 |
| Booster + Orbiter | --- | --- | 811,000 | --- |

## 3.0 RECOVERY CONCEPTS

Early in this study an attempt was made to identify all reasonable approaches to recovery of an orbiter vehicle after rendezvous with an airplane or booster carrier vehicle. Advantages and disadvantages were listed for each concept. The most promising of these concepts were selected for study in more detail, based on judgement evaluation of these lists. The selected concepts, which are described in detail in section 4.0, are:

(1) Docking of orbiter and airplane.

(2) Docking of orbiter and booster

(3) Towing of the orbiter by the airplane until the orbiter is released on final approach or is released at touchdown on the runway.

Table 4 lists the other concepts considered and the major problems that prevented further study of these approaches. Appendix A provides a description of these concepts and a listing of advantages and disadvantages.

TABLE 4

OTHER CONCEPTS CONSIDERED

| CONCEPTS | PROBLEMS |
|---|---|
| Towed Landing Package | Aerodynamic Interference |
| Airplane Tow, Circling Letdown | Not Applicable to Large Payloads |
| Parachute Landing | Poor Control of Landing Point |
| Parawing Landing | Development Incomplete, Large Size |
| Airplane Tow to Balloon Station | High Load Factor |
| Airplane Tow into Barrier | |
| Helicopter Landing | Payload too Heavy. |
| Sea Landing | Naval Support, Salt Water Environment, Transportation. |

## 4.0 RECOVERY SYSTEM CONCEPTUAL DESIGNS

The paragraphs which follow describe the mechanical implementation of the selected concepts including weights and vehicle modifications.

### 4.1 Docking

For the hard dock case, the carrier vehicle, whether an airplane or a booster which has been used to launch a second orbiter, has four position options relative to the orbiter available, namely, head-to-tail, tail-to-head, orbiter above and orbiter below.

Head-to-tail or tail-to-head hard dock results in rather adverse center of gravity positions for the composite vehicle rendering control in flight and/or landing extremely difficult if not impossible.

With the orbiter attached to the underside of the carrier vehicle, the longitudinal center of gravity problem is relieved, but the problem of getting the orbiter on the ground presents some difficulties. Landing with the orbiter aboard would require an extremely long landing gear on the carrier implying a new aircraft and probably eliminating the booster from consideration. It is conceivable that the orbiter could be released and allowed to land unpowered. Such an approach tends to negate some of the gains made, since the orbiter must land, requiring a landing gear. One remaining advantage is the range extension provided by the carrier. A further problem anticipated is due to the engines of an airplane located below its wing. The wing down wash and engine exhaust might create turbulence making the docking maneuver difficult.

Hard dock of the orbiter on the upper surface of the carrier appears to minimize the aerodynamic interference problem, the landing problem and center of gravity problem, and was selected as the hard dock position. It is felt that docking at this position is feasible with either a recoverable booster or an appropriately equipped aircraft.

#### 4.1.1 Orbiter-Airplane

In the case of docking the orbiter with an airplane, two basic methods of capture were considered. The airplane could support a cable to engage a hook on the under side of the orbiter, draw the orbiter to it, and bring into a hard dock. The second method is to extend a probe on a telescoping boom, which would engage a drogue or socket in the orbiter, and then maneuver the boom and the vehicle to a hard dock.

The first method was discarded in favor of the second because, in the first method, good control between the vehicles would be difficult to maintain when the cable length is small, final docking would probably require the extension of a boom to secure the orbiter, and in the event of a sudden loss of lift, the orbiter would not be constrained and could fall into the carrier vehicle.

In employing the probe and drogue concept for docking with the C-5A aircraft the following procedure is used (Refer to Figure 11):

(1) The orbiter is positioned within 50 feet and above the airplane. (Rendezvous guidance for accomplishing this is discussed in Section 7.0).

(2) The airplane establishes a parallel glide path with the orbiter at essentially the same velocity.

(3) An observer-pilot near the rear of the airplane controls and maneuvers the aircraft while a forward boom operator engages the socket in the orbiter with the telescoping gimballed boom.

(4) The probe, on engagement with the drogue or socket, locks in place. The drogue is gimballed and aligned with the orbiter center of gravity.

(5) The orbiter pilot decreases angle of attack slightly to maintain a nominal compression load on the boom.

(6) As the crafts approach each other, the rear observer raises a second boom to engage a drogue at the rear of the orbiter. The rear drogue is gimballed and has fore and aft freedom of movement to account for misalignment and movement of drogue due to flight effects such as airframe deflections and heating.

(7) The two booms now draw the orbiter to the final docked position and are locked. Two adjustable chocks (sway braces) straddling the rear boom are provided for roll stability.

(8) The airplane pulls out of the glide with the orbiter in place, flies to a predetermined base, and lands.

Figure 11 shows the C-5A in position to initiate and accomplish the docking and the major elements required to accomplish a hard dock with the orbiter. At engagement the vehicles are connected at or near their longitudinal centers-of-gravity, thereby minimizing any induced moments. Retraction of the forward boom is slowed or stopped when the vehicles are approximately ten feet apart, and the rear boom is engaged. Both actuators then draw the vehicles together at which time two chocking pads contact the orbiter at points near the aft boom. These pads are adjusted to a predetermined preload, thereby providing roll restraint. It should be noted that the forward boom is drawn inside the aircraft during the last ten feet by an extension mechanism, probably another hydraulic actuator as shown.

The insertion of the forward actuator is similar to the engagement of the air-to-air refueling boom between Air Force tankers and bombers. The

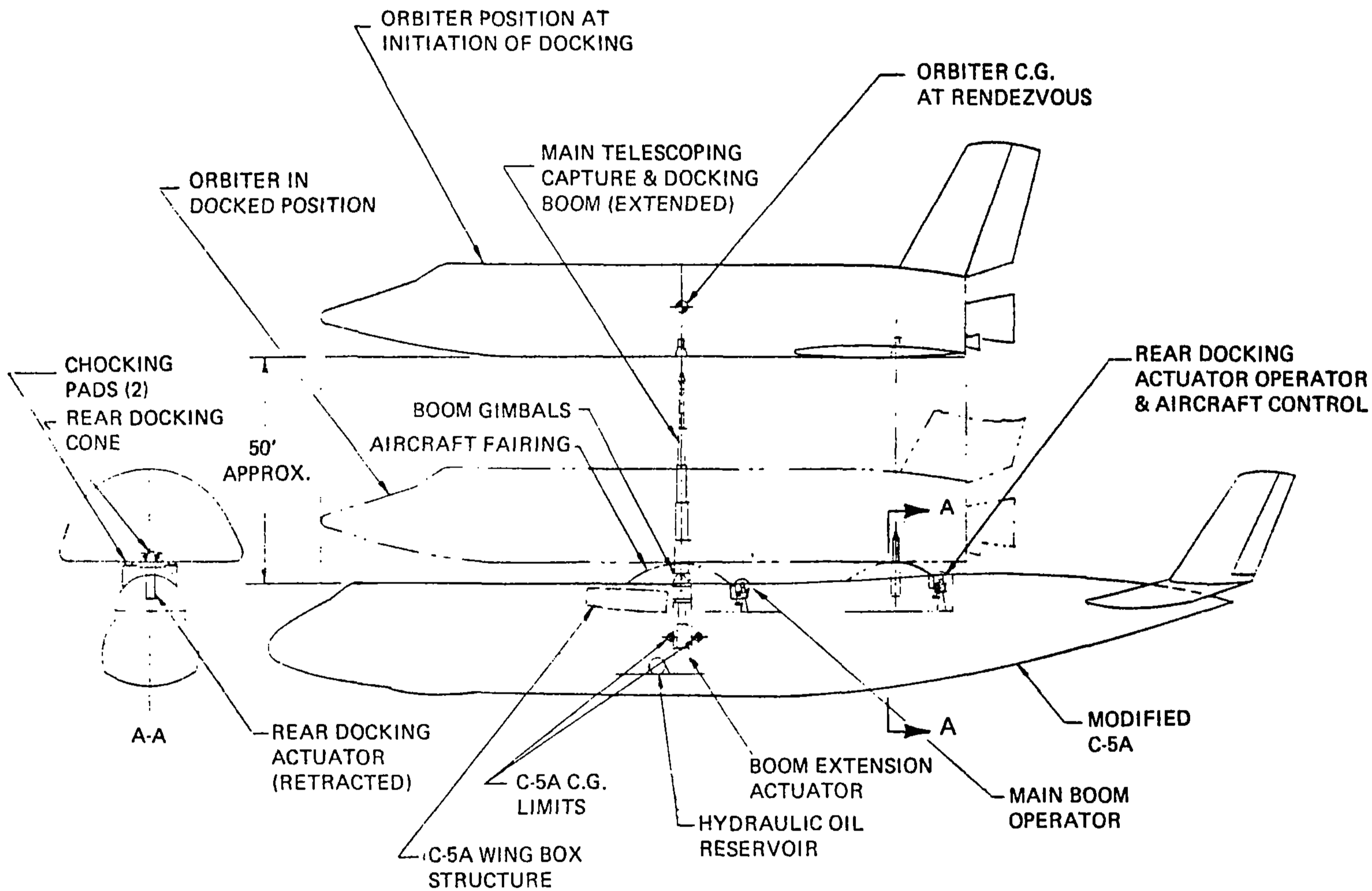



Figure 11. - Orbiter-Airplane Docking

separation distance is about the same (about 50 feet). Figure 12 shows a B-52 engaged in such a refueling maneuver. The refueling boom controlled by an operator in the tanker is gimballed and telescopes. Additional information and photographs may be found in Reference 5.

Figure 13 shows the forward telescoping boom in greater detail. The boom, a five segment hydraulic actuator, is capable of exerting force in either direction. The aft boom is similar but with only one or two segments. The forward boom is structurally capable of carrying a 400,000 pound load in either tension or compression. It is operated by the 3000 psi system of the aircraft at an operating pressure of 2800 psi. The C-5A has a considerable reserve of hydraulic capacity; therefore, hydraulic power was chosen as the prime power source for the docking system. An input flow rate of 70 gallons per minute extends the boom in a little over two minutes, and an input flow rate of 30 gallons per minute retracts the boom in one minute. A retraction flow rate of 70 gpm input to the extension actuator, completes the retraction into the fuselage in one minute. Lower flow rates are required for the aft boom.

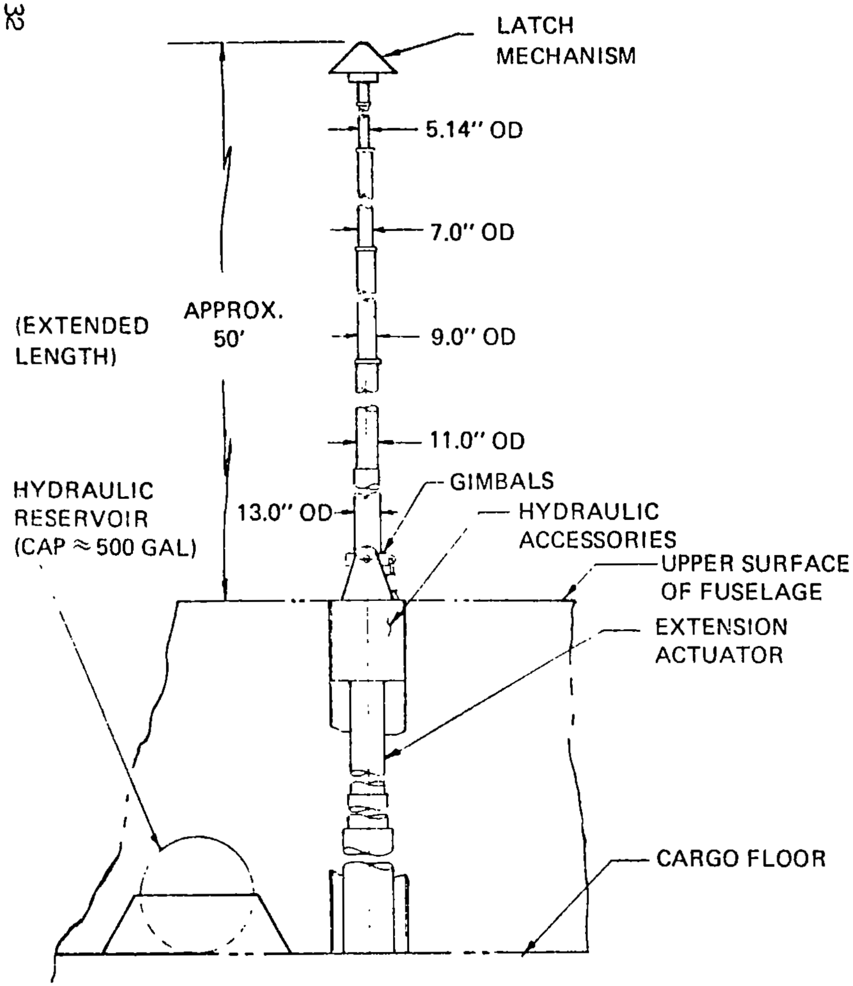
The boom when extended to the upper surface is gimballed in two planes with a freedom of $\pm 15^{\circ}$. The forward docking cone of the orbiter is also gimballed to give the same angular freedom of motion. This gimballing of the boom along with extension and retraction results in a three degrees-of-freedom system.

The hydraulic accessories section below the gimbal supports includes the valves, filters,accumulator and other components required for the boom hydraulics. The boom mechanism is mounted on the cargo floor along with a 500 gallon reservoir to augment the airplane system.

The forward operator, shown in Figure 11, controls the main boom during all operations. The rear boom is under control of the aft operator until both are locked into the orbiter docking cones. After this operation is complete both booms are slaved and operated together by the main boom operator. When fully retracted, the booms are locked in place to carry the flight loads. The forward boom carries vertical, fore and aft, and side loads. The aft boom carries vertical and side loads only. Freedom of motion in a longitudinal direction is permitted for the aft docking cone by means of tracks or slotted holes to take care of tolerances and to prevent fore and aft loads being induced in the aft boom. The chocking pads accept only down loads and loads induced by friction.

When the latching mechanism enters the docking cone and is aligned and fully engaged, it will automatically lock in position. Figure 14 is a sketch of such a mechanism. In the position shown the mechanism is aligned in the orbiter docking cone just prior to full engagement. As the boom upper section 1 moves up, the springs 2 inside the latch cone 3 (3 segment cone) are depressed and the boom section moves out of the cone. As the lock pins 4 (3) are aligned with the lock groove 5, the lock plunger 6 is driven upward by a compression spring causing the pins to be

Figure 12. - Air-to-Air Refueling of B-52 (Courtesy Air Force Museum, Wright-Patterson Air Force Base)




HYDRAULIC CONTROL:

- EXTEND-RETRACT SECTIONS.
- TWO-GIMBAL DRIVE AT BASE.
- EXTEND-RETRACT FROM AIRPLANE.

DESIGN LOAD – 400,000 LBS

| HYDRAULIC SYSTEM WEIGHT | |
|---|---|
| TELESCOPING BOOM | 3400 LBS |
| GIMBAL SYSTEM | 400 LBS |
| REAR BOOM | 1000 LBS |
| EXTENSION SYSTEM | 3500 LBS |
| ACCESSORIES & PLUMBING | 700 LBS |
| HYDRAULIC FLUID | 3000 LBS |
| TOTAL | 12,000 LBS |

Figure 13. Telescoping Boom

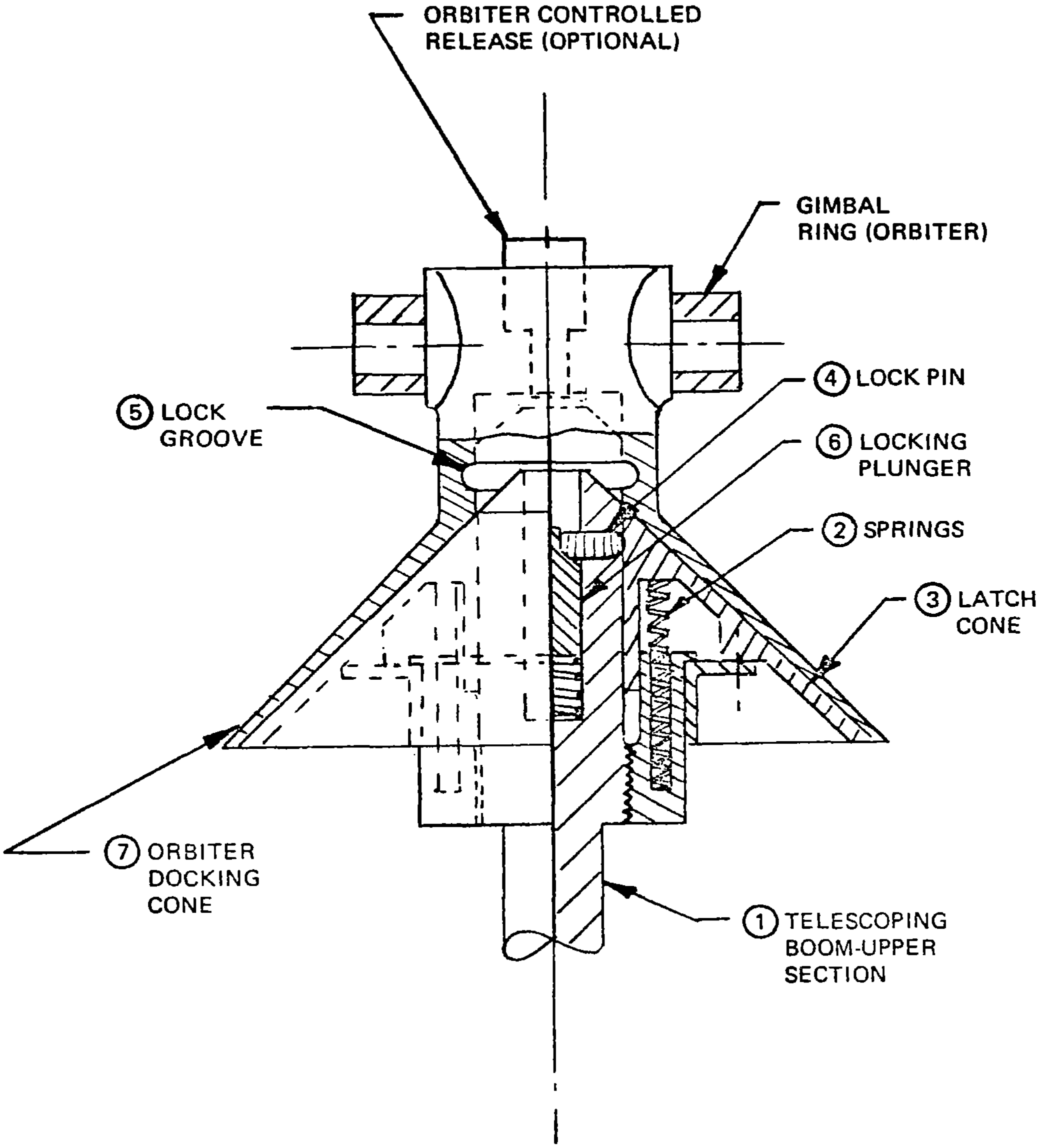



Figure 14. - Latch Mechanism

forced into the groove thereby locking the upper boom section into the docking cone. A release can be incorporated in the docking cone to drive the locking plunger down, releasing the pins and permitting the boom upper section to withdraw from the docking cone.

Figure 15 shows a typical telescoping boom joint which might be used. To extend the sections fluid fills the boom from the bottom, expelling fluid from between the sections. To retract the boom, fluid passes through hydraulic lines in the section walls forcing fluid out through a passage at the bottom of the boom as the section moves.

Pressure relief devices will be placed in each section passage and the bottom of the boom to prevent overpressurization and to absorb sudden overloads on the boom.

Weight

Returning to Figure 13, a table on this drawing shows a breakdown of the major weights for the hydraulic system. These total approximately 12,000 pounds. This weight coupled with the orbiter weight of 200,000 pounds leaves approximately 50,000 lbs for additional modification to the airplane.

Other Airplane Modifications

In addition to the two new crew stations, and the docking system (including hydraulics), other modifications are required. A major modification of the upper fuselage structure and beef up of the cargo floor will be required to carry the additional loads imposed on the aircraft. A high drag device, which is discussed in Section 5.3, must be added. The high tail configuration of the C-5A will probably be ineffective with the orbiter on the fuselage upper surface and, as indicated in Figure 11, a low horizontal and twin verticals will probably be required.

### 4.1.2 Orbiter-Booster

Figure 16 shows the approximate relative positions of the orbiter and booster for a hard dock. The implementation would be basically the same as that described for the airplane.

Additional booster requirements to provide a docking capability are:

(1) The docking equipment.

(2) Two crew stations, windows and controls.

(3) Support structure for docking equipment and crew stations.

(4) Propulsion capability during the docking operation (200,000 pounds thrust, 27,000 pounds of fuel per minute). This is discussed further in Section 5.2.

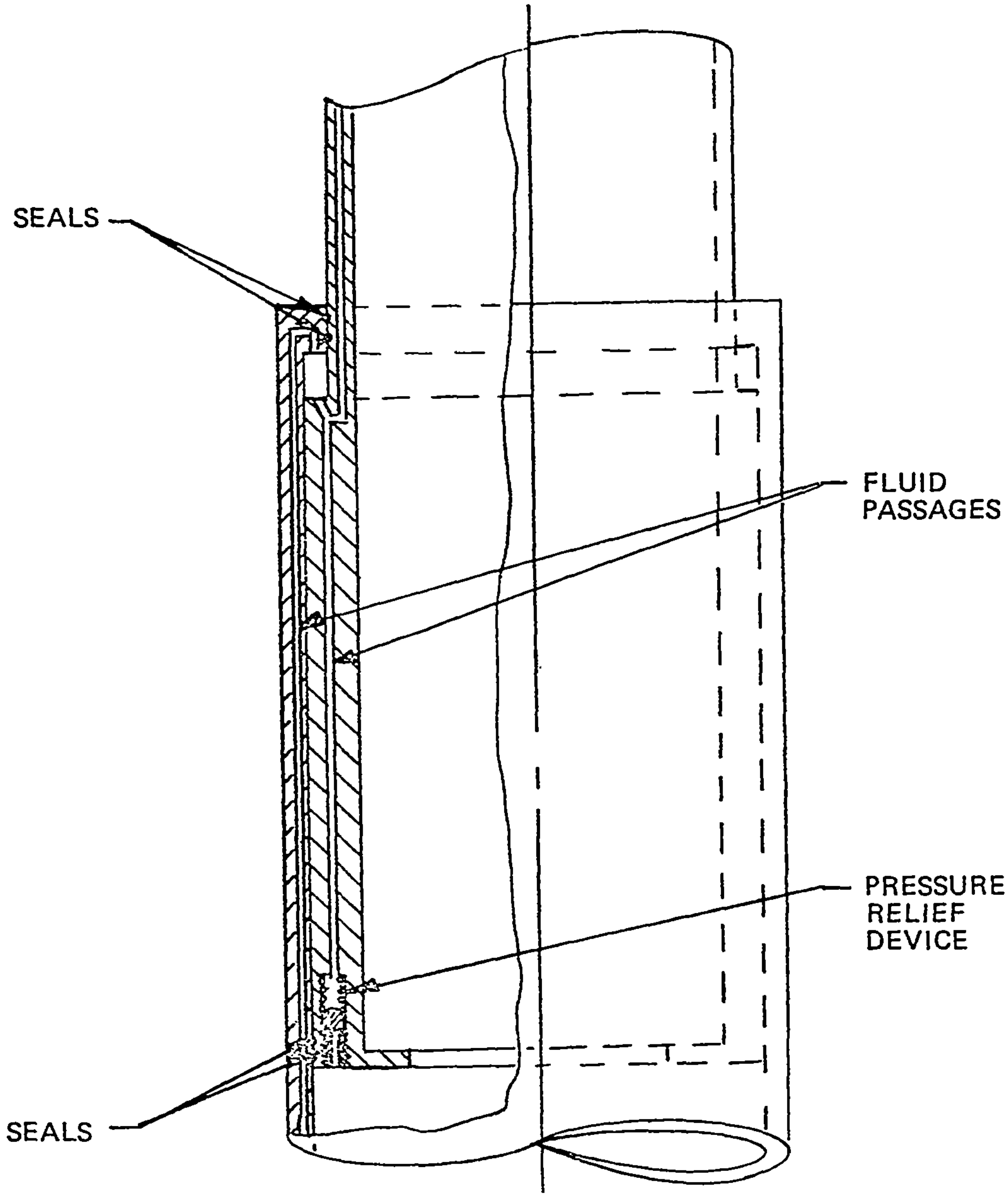



Figure 15. - Typical Joint, Telescoping Boom

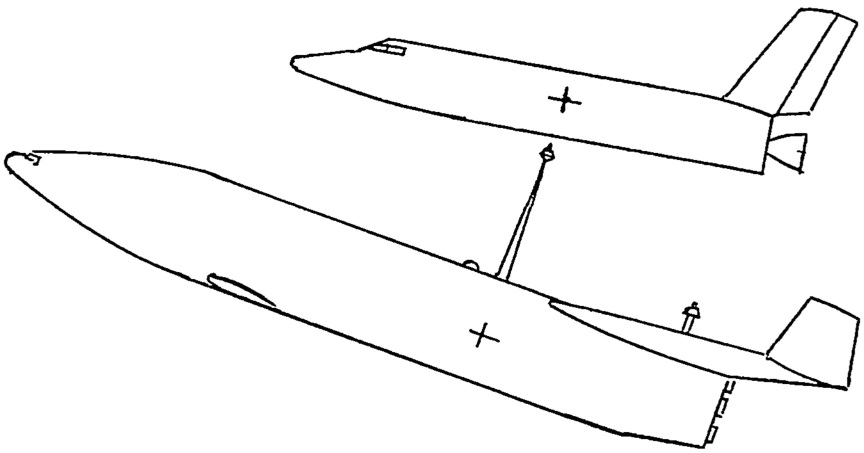


Figure 16. - Orbiter-Booster Docking

(5) Thermal protection for the upper fuselage, crew stations, telescoping boom and latch (See Section 4.1.6).

### 4.1.3 Docking Dynamics

Each of the two docking vehicles and the coupled combination must by dynamically stable and controllable during the entire docking operation. Analysis of this operation is a complex problem. At least twelve modes of motion should be considered, including;

(1) Three longitudinal degrees-of-freedom for the orbiter.

(2) Six degrees-of-freedom for the airplane

(3) Three degrees of freedom for the main telescoping boom.

An even more thorough analysis would include dynamics of the smaller boom on the carrier vehicle and the gimballed docking cones on the orbiter. Aerodynamic interference must be considered, including variations with separation distance between the vehicles. Flexibility of the boom may be important, and should be considered. The analysis should also include all significant dynamic characteristics of both vehicles and the docking system, including control logic with optimized system gains.

Although an analysis of this depth could not be conducted within the scope of this study, the conceptual design defined in Section 4.1.1 is provided with several features that should minimize the probability of serious design problems. These are:

(1) Initial attachment occurs with vehicles separated by almost fifty feet to minimize aerodynamic disturbances. Relative position of the vehicles appears to be more favorable than that for air-to-air refueling.

(2) The main boom and docking cone are located so that loads are applied close to the center-of-gravity of both vehicles to minimize rotational disturbances.

(3) The capability to make continuous control inputs by crew members or autopilots should provide effective control of alignment throughout the docking operation.

A dynamic analysis of the type described above is highly recommended to be included in any further study of atmospheric rendezvous operations.

### 4.1.4 Docking Forces

Impulse required and energy absorbed in initial docking maneuvers are functions of vehicle masses and relative velocity. Parametric data are shown in Figure 17 for relative velocities up to five feet per second.

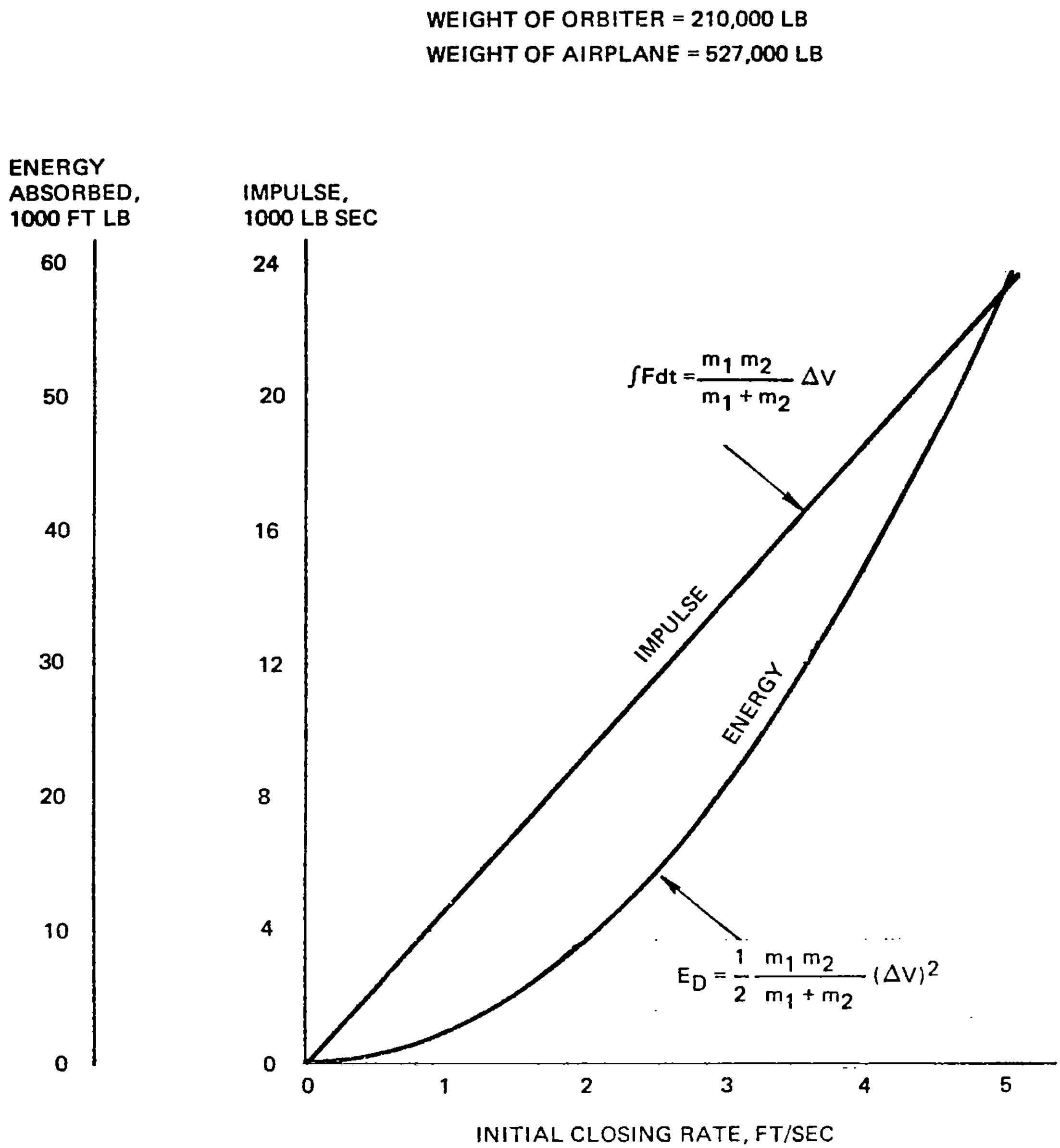



Figure 17. - Impulse and Energy Required for Docking

Closing rates well within this range should be possible based on experience such as formation flying, air-to-air refueling, and docking of spacecraft. The hydraulic system described in Section 4.1.1 can be designed to pressures and flow rates corresponding to docking velocities in excess of five feet per second.

### 4.1.5 Effects of Air Turbulence

The degree of difficulty and time required to complete the orbiter-airplane docking operation are probably significantly increased under conditions of severe air turbulence. The primary approach to solving this problem is to select a rendezvous area where weather conditions are as favorable as possible. The excellent lateral range capability for this type of recovery (See Section 5.4) should provide an enormous area from which a rendezvous location can be selected. Time of rendezvous can also be selected, within operational limitations, to provide favorable weather, except for some emergency conditions.

Clear air turbulence, which apparently is not predictable, generally occurs at altitudes above the estimated altitude for initiation of docking (32,000 feet). If excessive turbulence is encountered, a brief delay to descend to better conditions at a lower altitude can be tolerated (See Section 6.1).

In general, the capabilities to select timing and favorable weather conditions and to achieve initial latching are all considered better for the conceptual docking design than for an operational air-to-air refueling.

### 4.1.6 Aerodynamic Heating

The purpose of the thermal analysis was to evaluate the potential problems that might occur during the rendezvous and latch-up phases due to elevated temperatures on the orbiter surface and structure from re-entry heating. Emphasis was placed on subsonic rendezvous with an aircraft, although orbiter temperatures were also determined for the time period when hypersonic rendezvous would occur. Specific problems considered were heating effects on the top surface of the aircraft, thermal protection requirements for the aircraft crew members at stations on the top of the aircraft, and temperature effects on the docking boom and latching mechanism.

Temperatures for the lower surface of the orbiter as a function of time from a subsonic rendezvous and latchup are shown in Figure 18. Heating rate histories for the curves of Figure 18 were based on a nominal trajectory for establishing testing environments for the orbiter fuselage panels, including cold wall heating rates, stagnation temperatures, and local pressures as a function of time. These data were used as input in a thermal analysis to determine both surface and substructure temperatures at three locations on the bottom centerline of the orbiter, as indicated in the sketch in Figure 18.

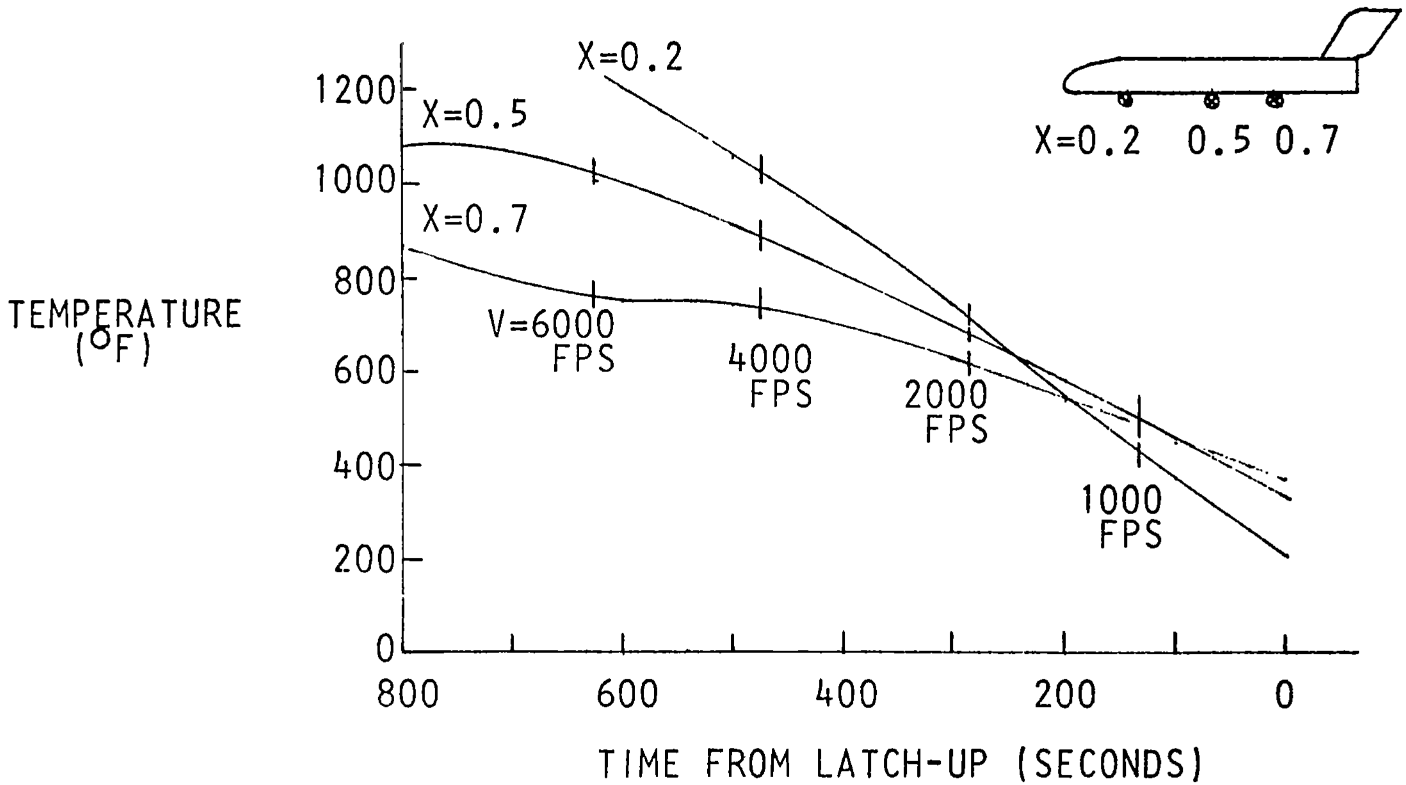



Figure 18. - Orbiter Lower Surface Temperatures

The temperature profiles between the surface and the substructure at the time of peak heating and at the time of subsonic rendezvous are shown in Figure 19. Also shown in Figure 19 is a sketch of the assumed configuration of the orbiter lower surface insulation and structure. The Zr REI (re-entry insulation) material and the thermal properties assumed in the analysis were based on information in Reference 6.

Figures 18 and 19 show that although peak surface temperatures in excess of 1500°F will be reached on the bottom surface of the orbiter during high re-entry heating, all lower surface temperatures are less than 500°F at the time of a subsonic rendezvous. Further, these surface temperatures are decreasing rapidly because of the low heating and effective aerodynamic cooling at the lower altitudes. Figure 20 shows the effect of radiant heating from the orbiter on the top surface of the aircraft. An increase in top-surface temperature of less than 25°F would be expected. The dominant thermal control on both surfaces is the air flow between them, which tends to cool both surfaces. As the orbiter and aircraft are brought together by retraction of the boom, this air flow will be restricted and the thermal interchange between the orbiter and the aircraft becomes more complex. This will have very little effect for subsonic rendezvous, but could be more significant at hypersonic speeds.

The subsonic curves in Figure 19 show the thermal gradients that might be encountered through the insulation and substructure at time of latch up. Temperatures of the aluminum structure will be less than 300°F, but temperatures close to 800°F could be encountered at points inside the insulation. Temperatures on the boom itself, however, will remain low, since the cooling effects from the air flow will dominate during the rendezvous and latchup phase. No thermal protection for the crew should be required at the indicated surface temperatures.

The thermal problems for a hypersonic rendezvous have not been analyzed in detail, but they would be more severe in several respects. As shown in Figure 18, orbiter lower surface temperatures at 6000 feet per second are in the range of 1000-1200°F. Although this should not present a problem to the top surface of the booster, some type of thermal protection would be required for crew members controlling the latchup operation. High temperatures will also be encountered on the boom from aerodynamic heating. An additional problem could be structural heating on the orbiter, since cooling of the lower surface will be inhibited by restricted air flow after latchup.

4.2 Towing

4.2.1 Orbiter - Airplane

The sequence of events for the capture and retrieval of the orbiter by the C-5A using a towing concept is as follows (Refer to Figure 21):

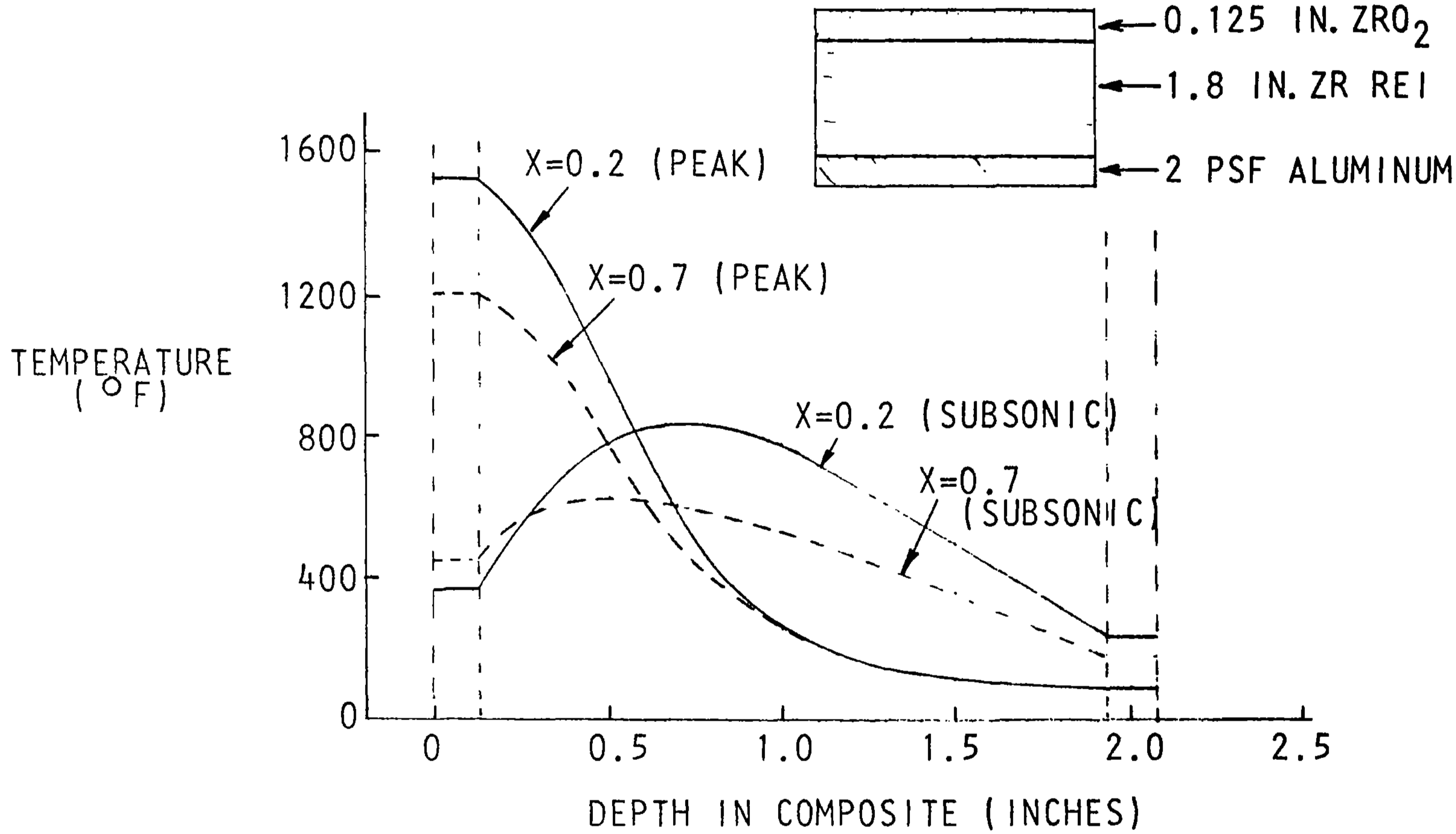



Figure 19. - Lower Surface Temperature Profiles

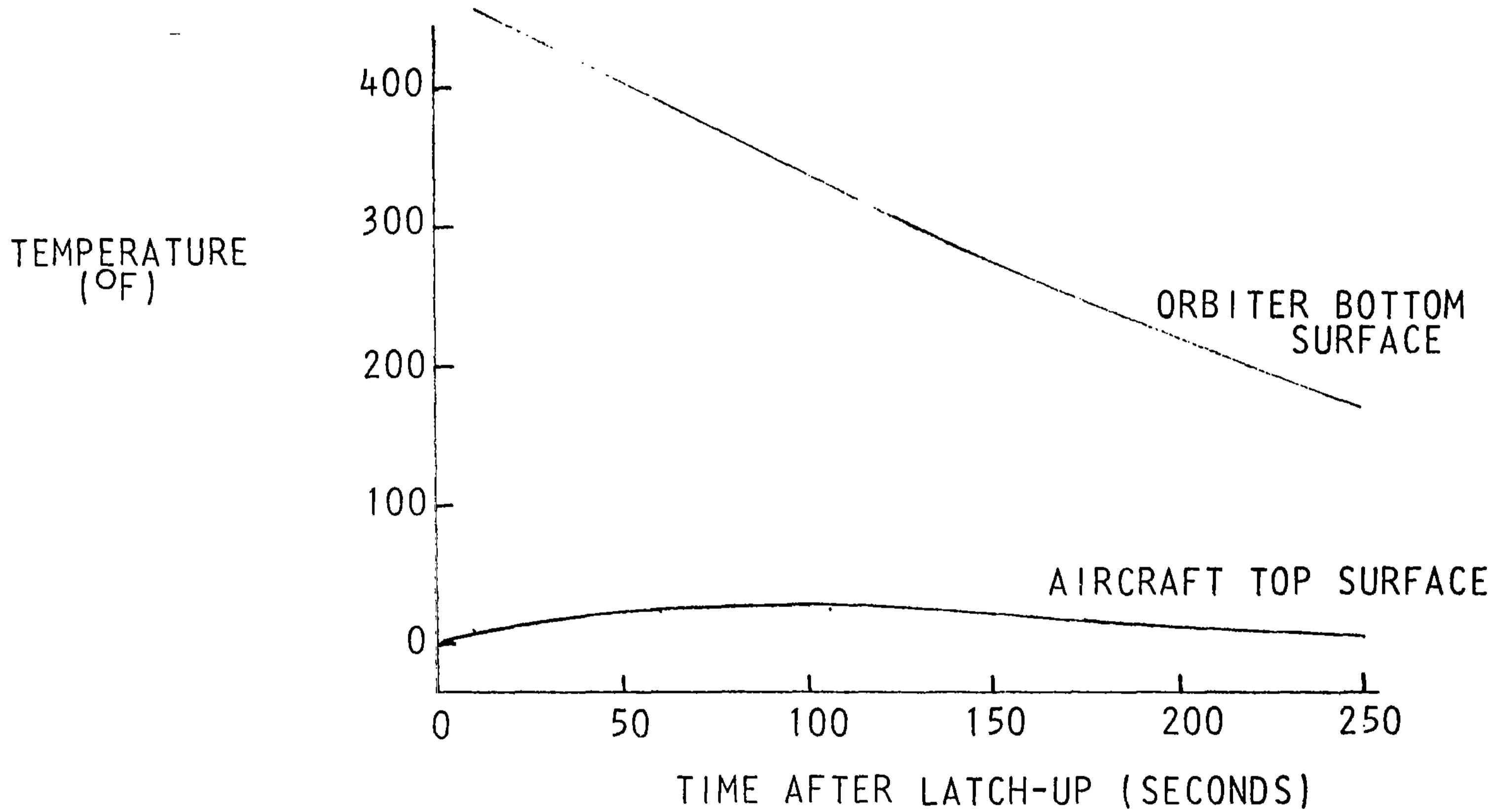



Figure 20. - Heating of Aircraft during Docking

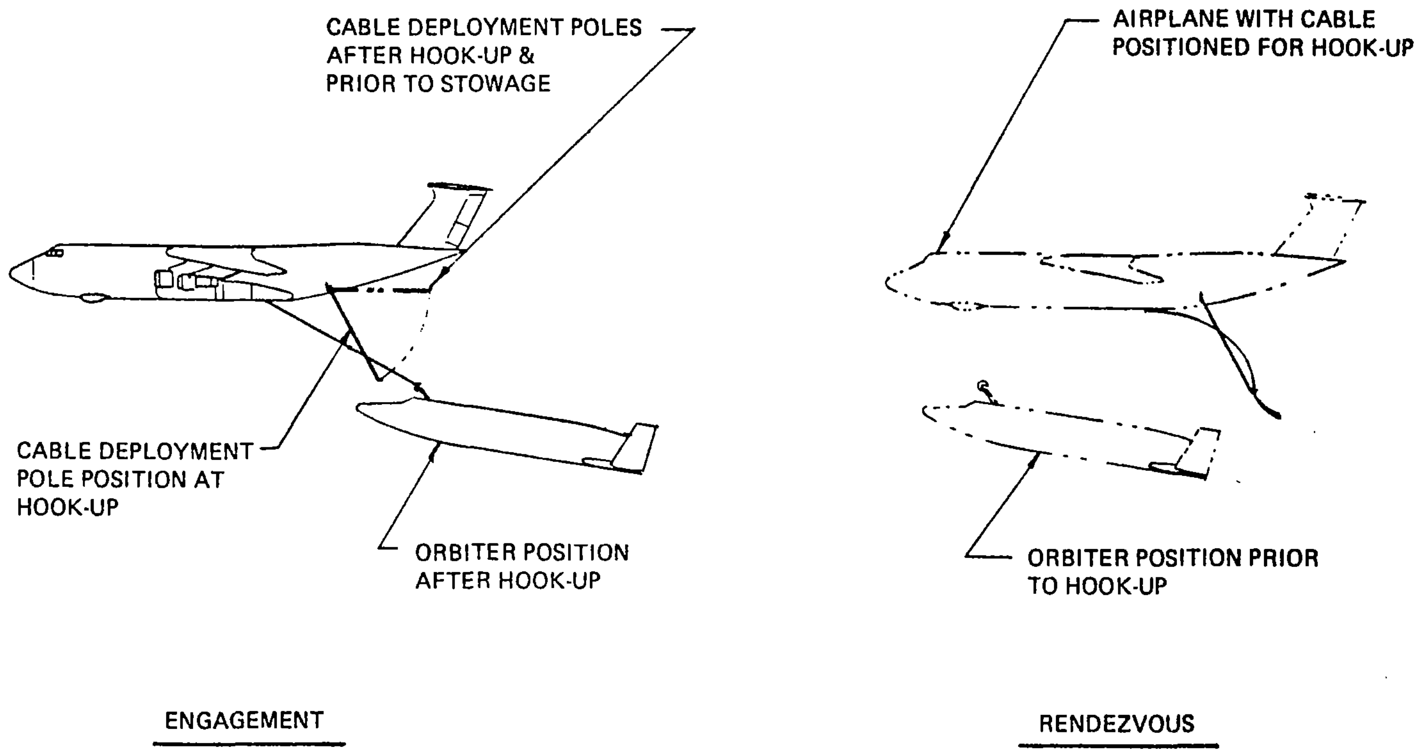



Figure 21. - Airplane Tow

(1) The orbiter is permitted to descend ahead of the airplane.

(2) The airplane accelerates, passing about 50 feet above the orbiter at a relatively slow relative velocity.

(3) A cable loop displaced below the C-5A on poles controlled by an observer in the C-5A is allowed to pass near the upper surface of the orbiter and engage a hook extended upward from the orbiter.

(4) The cable on engagement with the orbiter hook is locked in place and automatically released from the deployment poles.

(5) The deployment poles are retracted to a position along the airplane body.

(6) The airplane pulls out of its glide towing the orbiter with it.

(7) The orbiter adjusts angle-of-attack to control the cable tow angle such that cable tension forces act near the center-of-gravity of both vehicles.

(8) The airplane cruises to predetermined landing site and reels out cable until approximately 1000 feet of cable are deployed.

(9) The airplane tows orbiter to an essentially zero sink rate landing.

(10) At touchdown, the orbiter releases the tow cable and rolls to a stop using brakes and drag chute to decelerate.

(11) The aircraft reels in cable while circling.

(12) The aircraft makes a normal approach and landing.

Orbiter weight at rendezvous is slightly greater than for the docking case due to the requirement for a landing gear. Weight probably increases by about three percent, or 6000 pounds.

Figure 21 shows the airplane and orbiter positions at rendezvous and shortly after the cable has been engaged by the orbiter hook. The separation distance between the vehicles is approximately 50 feet and the relative velocity quite low (5 to 10 feet per second). The right hand sketch shows the cable loop deployed on poles and the orbiter hook raised to accept it. The orbiter in this illustration has twin vertical tails to avoid interference with the cable loop. The left hand sketch shows the cable engaged and the poles

retracted. The hook is positioned as close as possible to the orbiter center-of-gravity to minimize rotational perturbations due to the cable. It would be desirable, from a stability and control standpoint, to locate the hook even farther aft; however, this is limited in this baseline configuration by location of the payload bay. The line of action of the cable also passes near the airplane center-of-gravity to minimize moments imposed on the aircraft by the cable.

Equilibrium tow angle is shown in Figure 22 , as a function of equivalent airspeed and orbiter angle-of-attack. Tow angle is zero when the orbiter is directly behind the airplane. Conditions for which tension in the cable produces no pitching moment on the orbiter are shown by the dashed line. The orbiter can nominally operate near this line, and then vary angle-of-attack to control tow angle (or relative altitude with respect to the airplane). This indicates the capability to land the orbiter in the towed condition. The airplane could fly over the runway at an altitude of 500 to 600 feet, with a cable deployment of 1000 feet. Orbiter capability to control relative altitude permits it to approach and land at a very small sink rate and retain a go-around capability to the time of cable release at touchdown. For a dead-stick landing, the orbiter's sink rate at approach would be about eighty ft/sec, and the landing speed would be about twelve knots faster than a towed landing at the same angle-of-attack. Equilibrium cable tension for towing the 200,000 pound orbiter is shown in Figure 23.

Figure 24 shows the basic elements of the towing system in the airplane. The system is hydraulic powered to utilize the C-5A hydraulic capabilities.

The deployment booms are normally stowed at a minimized drag position and hold the cable loop against the aircraft body. The cable is guided from the winch and restrained in grooves along the airplane fuselage. The deployment booms are on a single shaft driven by a hydraulic motor through a gear box. A brief analysis has shown that a 16 horsepower motor is capable of deploying the booms against the air load in a few seconds. An alternate method of deploying the booms would be to use a rotary hydraulic actuator which might prove to be a lighter and simpler installation. The motion of the booms is controlled by the operator who can observe the deployment through transparent panels provided in the cargo floor and the outside contour of the aircraft. These windows permit the operator, who also controls the winch, to observe the hook up between the orbiter and the cable loop.

The cable is guided from the aircraft through a teflon lined tube to avoid the necessity of using large diameter pulleys or sheaves. The cable loads have been estimated at a maximum of 80,000 pounds which requires a fiberglass cable about 1-5/8" in diameter. Use of fiberglass avoids chafing in the event the loop should be dragged along the orbiter body prior to engagement. It is not known if fiberglass cable of this size is available. However, fiberglass cables over one inch in diameter have been used in tethered balloon operations, so no problem is anticipated in obtaining such a cable. A flexible

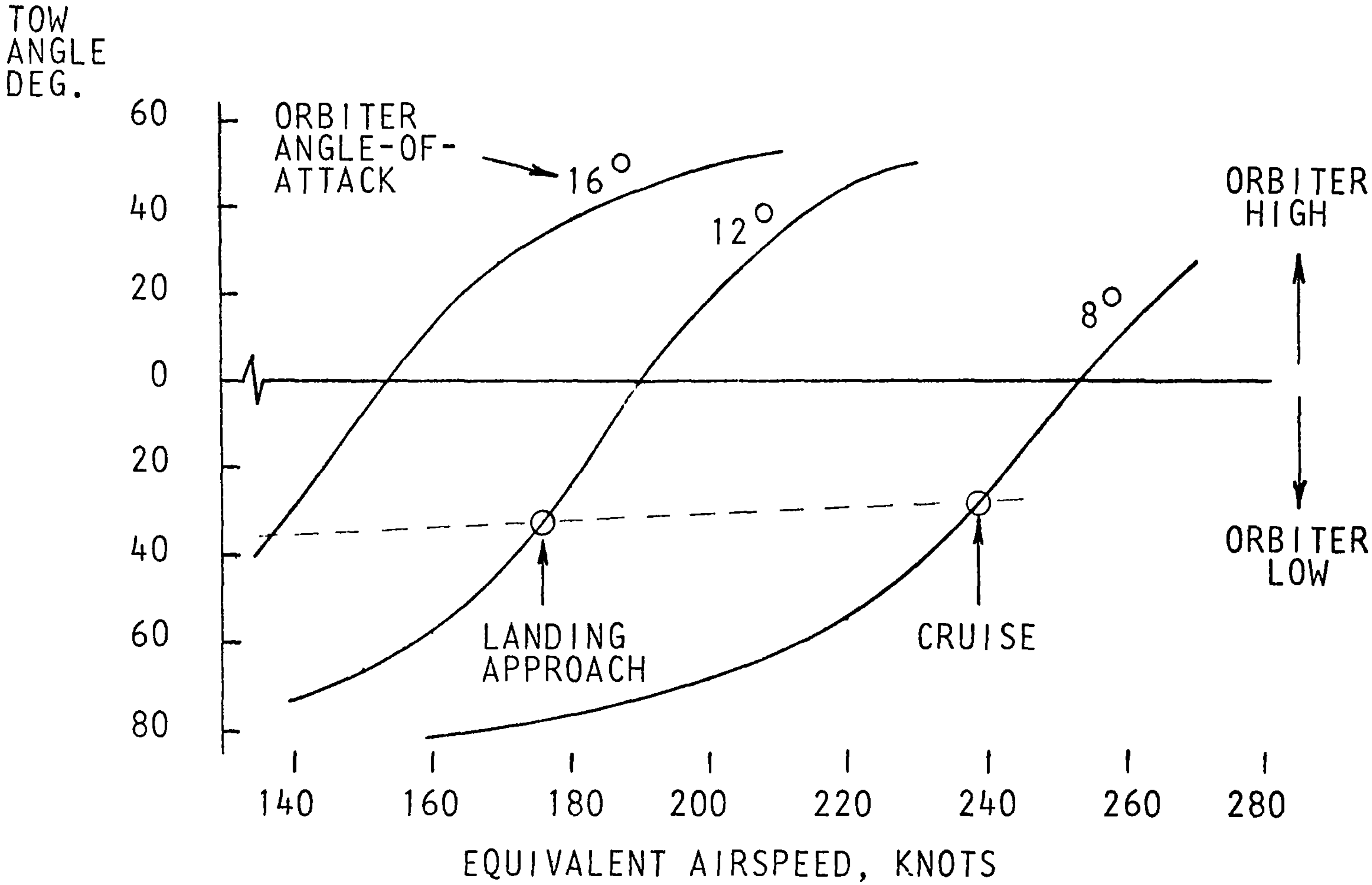



Figure 22. - Towing Characteristics

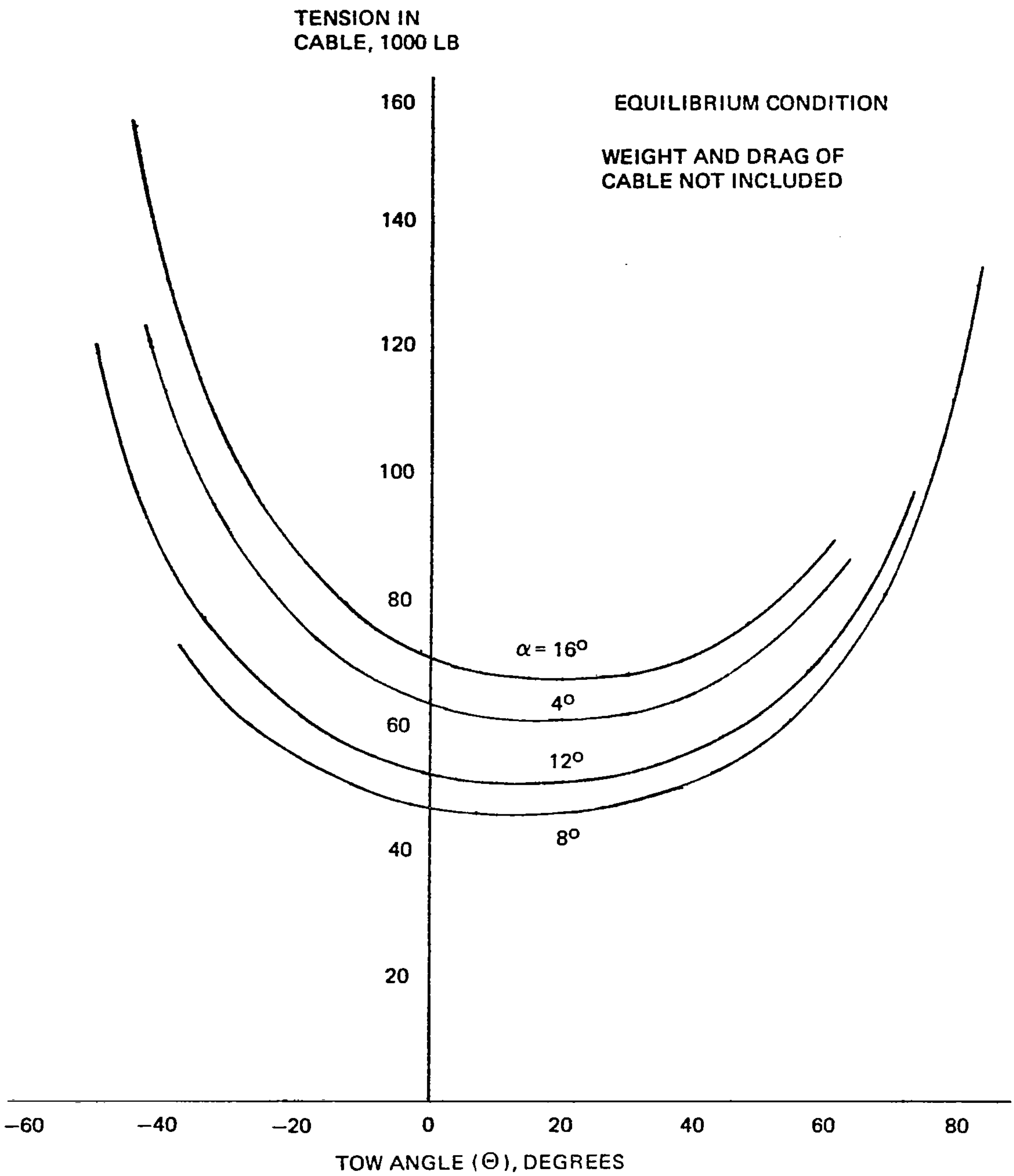



Figure 23. - Tension in Cable

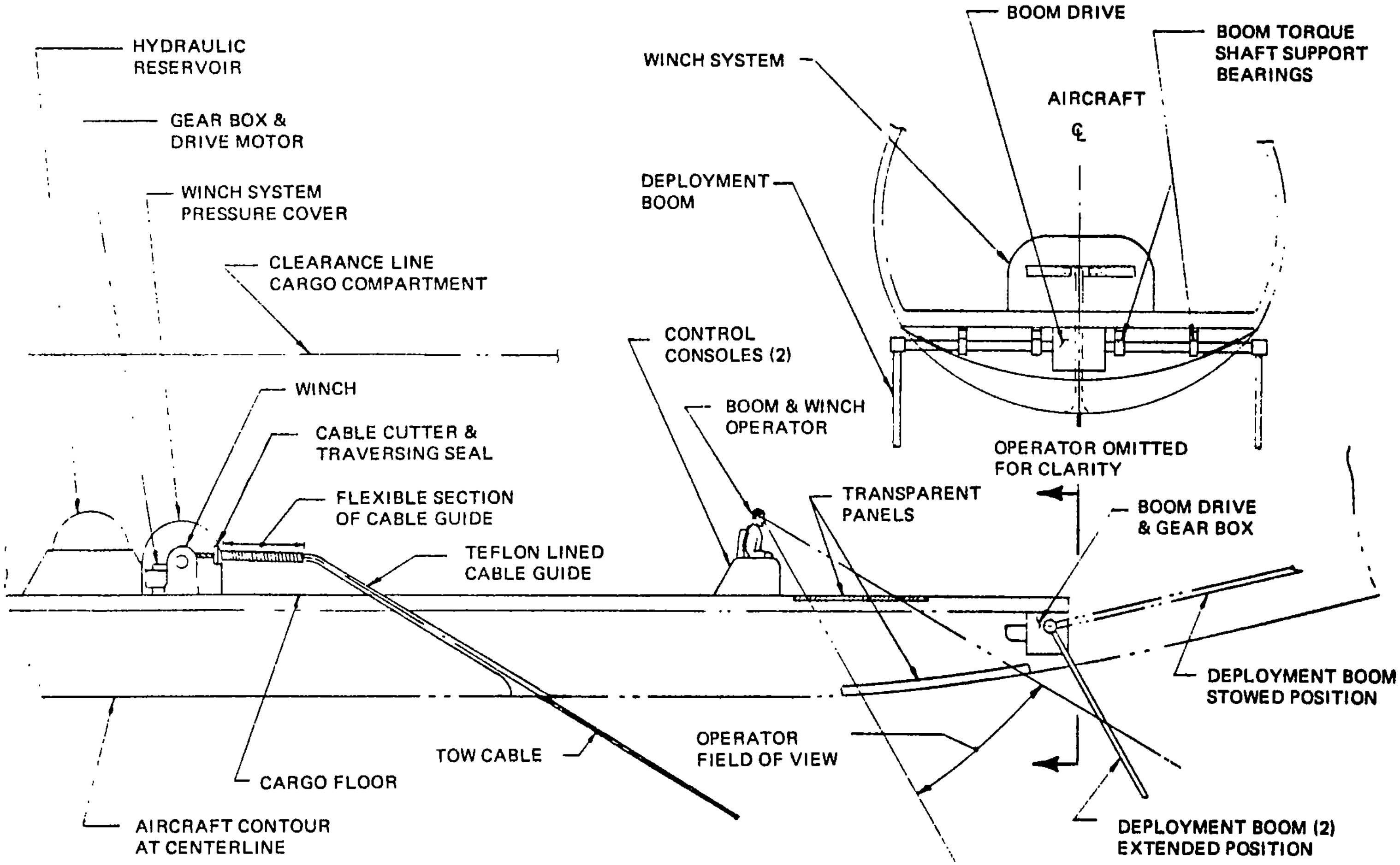



Figure 24. - Tow Mechanism Schematic

section and traversing seal have been included to permit proper wrapping on the winch drum. The winch is driven by a 180 horse-power hydraulic motor through a gear reduction box. This provides sufficient power to reel the cable in at about one foot per second under maximum load. A fluid flow rate of about 120 gallons per minute is required for this condition and is well within the C-5A pumping capacity. A hydraulic reservoir has been provided to augment the C-5A capacity. An accumulator, filters, valves, and plumbing complete the hydraulic system.

A pressure cover encloses the winch system, and the cable guide is connected to it through a traversing seal to maintain the pressure integrity of the cargo compartment.

At touchdown the orbiter must release the cable. One method for accomplishing this, as well as locking the cable in place at engagement, is shown in Figure 25. A rotating hook is provided which is normally held in place by a sear to carry the tension loads in the cable. A spring loaded finger allows the cable to enter the hook but prevents its being disengaged. to release the cable, the sear is withdrawn from the hook upon an electrical signal from the orbiter. The sear can be either a pyrotechnic device or a solenoid. The withdrawal force required (approximately 15,000 lbs) indicates a pyrotechnic device would be more applicable. The orbiter hook would normally be housed in a fairing atop the orbiter crew compartment and raised by means of a hydraulic or electric actuator.

A cable cutter has been provided in the aircraft to sever the cable when it has been reeled in since the cable loop probably would not pass through the cable guide and could cause problems when landing the airplane. The cutter could also be used for emergency disconnection of the cable.

Most of the elements of the towing system such as the winch, motors, etc. are off-the-shelf commercial hardware and those which are not can be produced with no advance in state-of-the-art and little development.

Table 5 shows a weight estimate for elements in the tow system. This estimate includes some of the modifications required to the aircraft, notably the fuselage modifications and additional crew station. Other modifications required are to provide a high drag device (as discussed in Section 5.3) and to increase thrust available due to the large additional drag of the towed orbiter. Current thrust is sufficient for four-engine low-altitude towing. Uprated engines or an additional engine is required to provide for an engine failure. Since the vertical component of cable tension represents a relatively small cargo weight increment, additional weight should not be a problem for airplane modification.

### 4.2.2 Booster-Airplane

It is shown in Section 2.3 that an additional reduction in booster size can be realized if the booster is recovered by an airplane. This permits removal of booster cruise engines and cruise fuel and a subsequent reduction

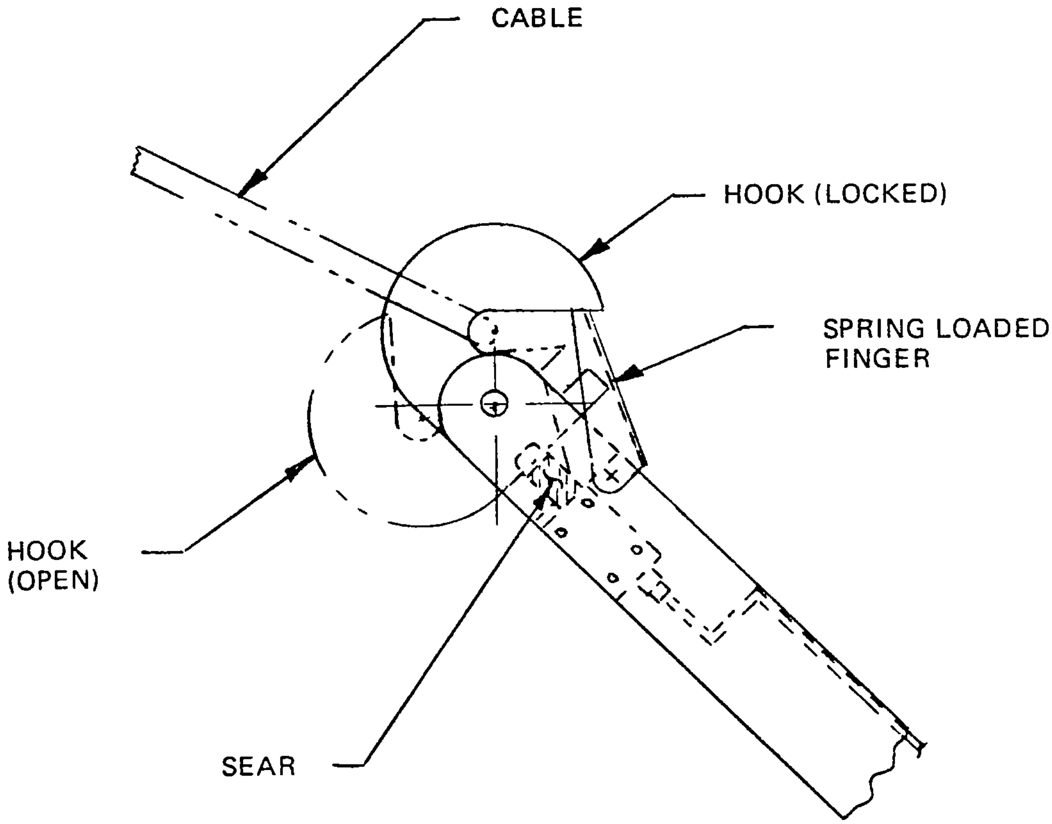



Figure 25. - Orbiter Hook Mechanism

TABLE 5

TOW CONCEPT WEIGHTS

| WEIGHT ADDED TO AIRCRAFT, LB: | |
| --- | --- |
| Support Structure | 1000 |
| Winch | 3000 |
| Gear Box | 1000 |
| Hydraulic Motor | 300 |
| Reservoir | 300 |
| Accumulator | 125 |
| Valves and Plumbing | 50 |
| Pressurization Covers | 300 |
| Controls | 250 |
| Observation Window | 400 |
| Deployment System | 3800 |
| Hydraulic Oil | 2100 |
| Cable | 1000 |
| Cable Guides | 100 |
| TOTAL | 13,725 |

Figure 21 shows that the orbiter cable hook can be located well aft of the nose to minimize distance between the cable attach point and center-of-gravity. The orbiter cargo bay prevents location of the hook closer to the c.g.; however, it is felt that further design studies would identify no serious stability problem due to the capability to augment stability with control inputs by the orbiter pilot or an autopilot.

Tow Cable Release

Energy stored in the cable due to the large tension was considered a possible hazard to the airplane or nearby objects on the ground when the cable is suddenly released at orbiter touchdown. A preliminary analysis was made to investigate the motion of the cable after release. It was found that the lift component of aerodynamic normal force on the cable causes it to rise. Drag prevents it from moving very far forward with respect to the airplane. It was concluded that this will probably not be a problem. Should further study indicate otherwise, possible fixes include:

(1) Installation of a high drag device near the end of the cable.

(2) Reduction of cable tension by the winch operator just prior to cable release.

## 5.0 AERODYNAMIC AND PERFORMANCE DATA

### 5.1 Orbiter

Subsonic - Subsonic aerodynamic data were obtained for a free stream Mach number = 0.8 which was considered to be a typical subsonic speed. Lift and drag coefficients for the body alone were taken from Ref. 8. To provide satisfactory equilibrium glide performance, the body shape selected had a reasonably high subsonic and hypersonic maximum lift-to-drag ratio, $(L/D)_{max}$. The shape parameters which define the selected body are (Refs. 8 or 9):

$$\text{Elliptical Cross-Section, } \frac{\text{Major Axis}}{\text{Minor Axis}} = \frac{a}{b} = 3.0$$

Planform and Profile Shape, Power-Body Exponent = n = 0.25

The body alone is highly unstable, and a large horizontal tail surface is required to achieve longitudinal stability. Refs. 8 and 10 were used to determine trimmed lift and drag coefficients for a suitable tail. With the horizontal tail at 0° deflection, the orbiter exhibits near-neutral stability for angles of attack up to 14°, leaving sufficient tail deflection available for vehicle maneuverability.

Total vehicle lift and drag coefficients were obtained by adding the components for the body and trimmed horizontal tail. Orbiter L/D values were calculated from the resulting force coefficients. The coefficients are based on a reference area of 6350 $ft^2$. Estimated subsonic aerodynamic data for the orbiter are shown in Figures 26 through 28.

Hypersonic - Hypersonic aerodynamic coefficients for the body alone were estimated using Ref. 9 and modified Newtonian theory. Ref. 9 provided data over an angle of attack range of $0° \leq \alpha \leq 25°$ at a Mach number of 4.63, the largest value investigated. Since aerodynamic coefficients remain essentially unchanged at high velocities, these data could be considered representative of the hypersonic speed regime, but the $\alpha$ range is insufficient for the orbiter re-entry requirements. Therefore, additional hypersonic aerodynamic data for $\alpha > 25°$ were estimated using modified Newtonian theory. Results of these calculations for the body were matched to those from Ref. 9 at $\alpha = 25°$ to obtain body force coefficients. Total vehicle lift and drag coefficients were then determined by the same procedure used for subsonic estimates. Estimates indicate that the selected tail can trim the orbiter at all angles of attack encountered during the reentry glide. Estimated hypersonic aerodynamic data for the orbiter are shown in Figures 29 through 31.

### 5.2 Booster

Aerodynamic data for the booster are based on McDonnell-Douglas Phase B studies. The aerodynamic reference area is increased from 10,000

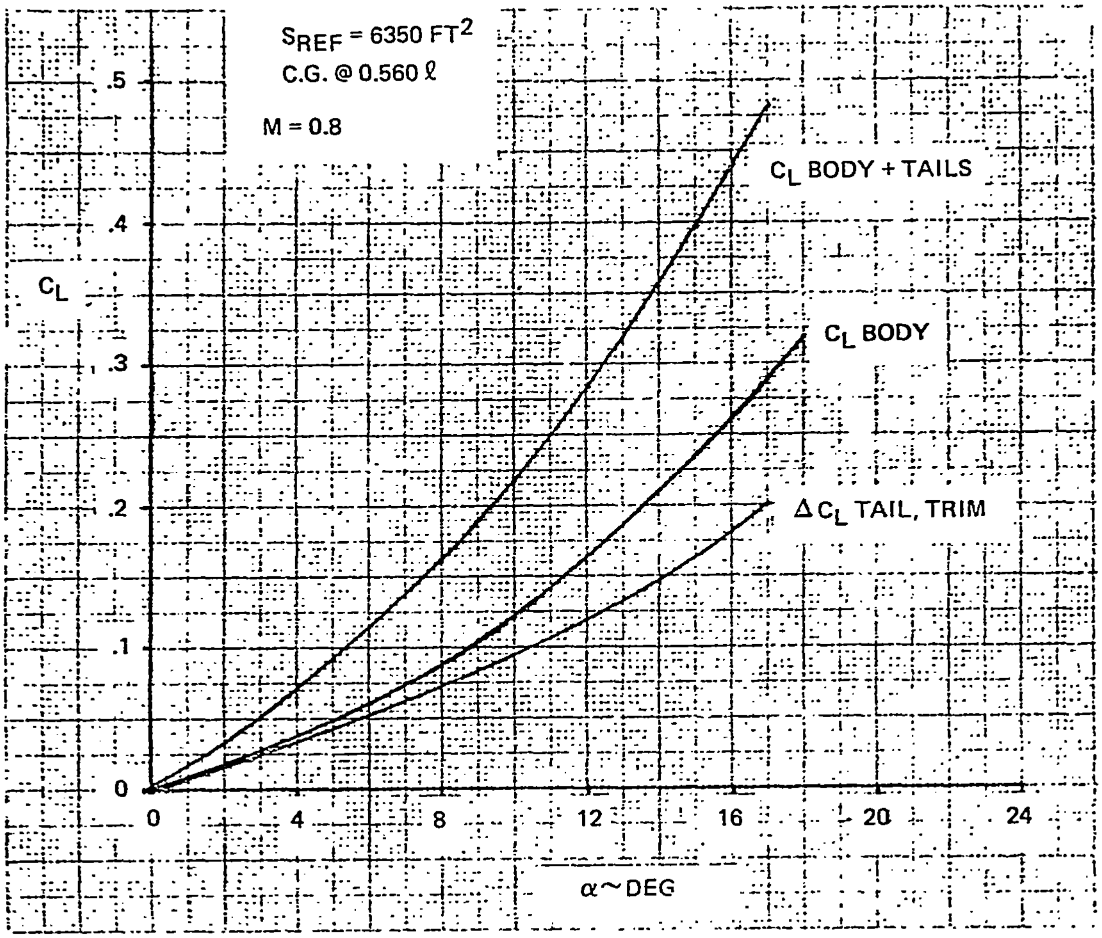



Figure 26. - Orbiter Subsonic Trimmed Lift Coefficient

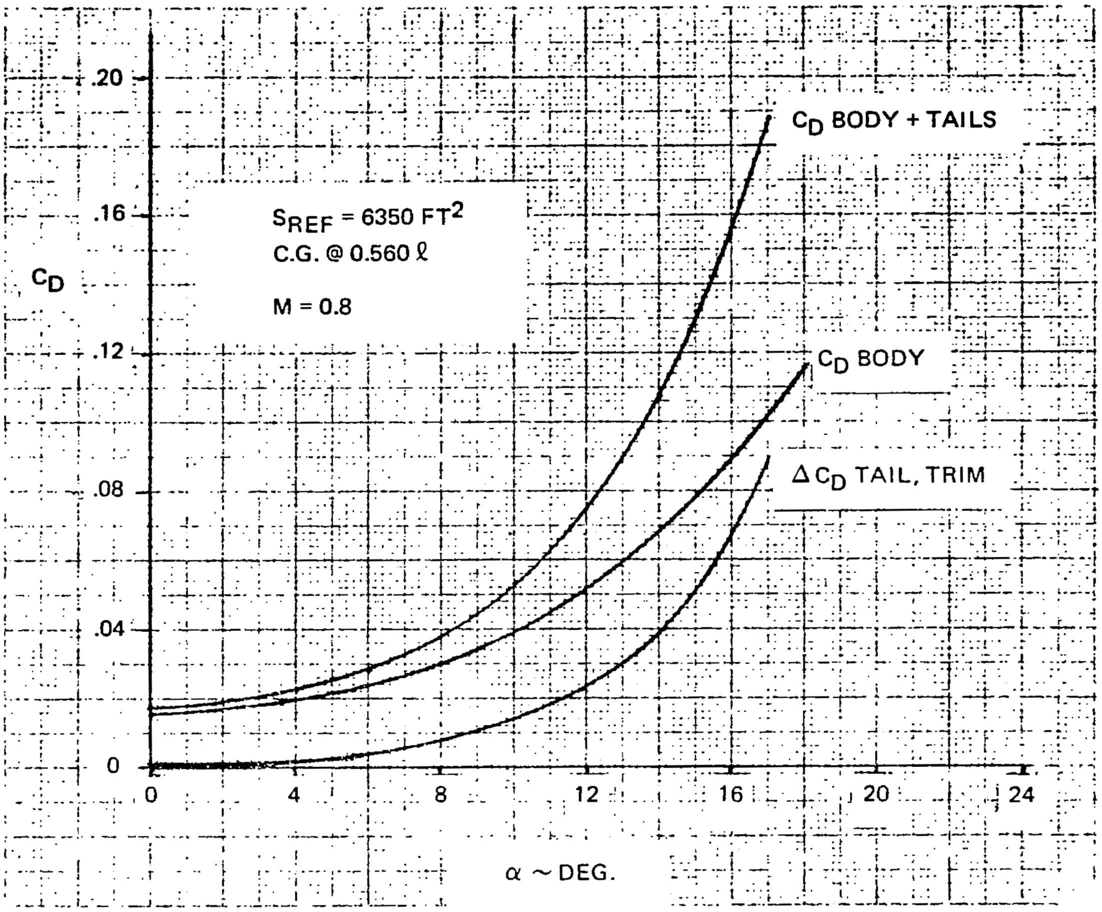



Figure 27. - Orbiter Subsonic Trimmed Drag Coefficient

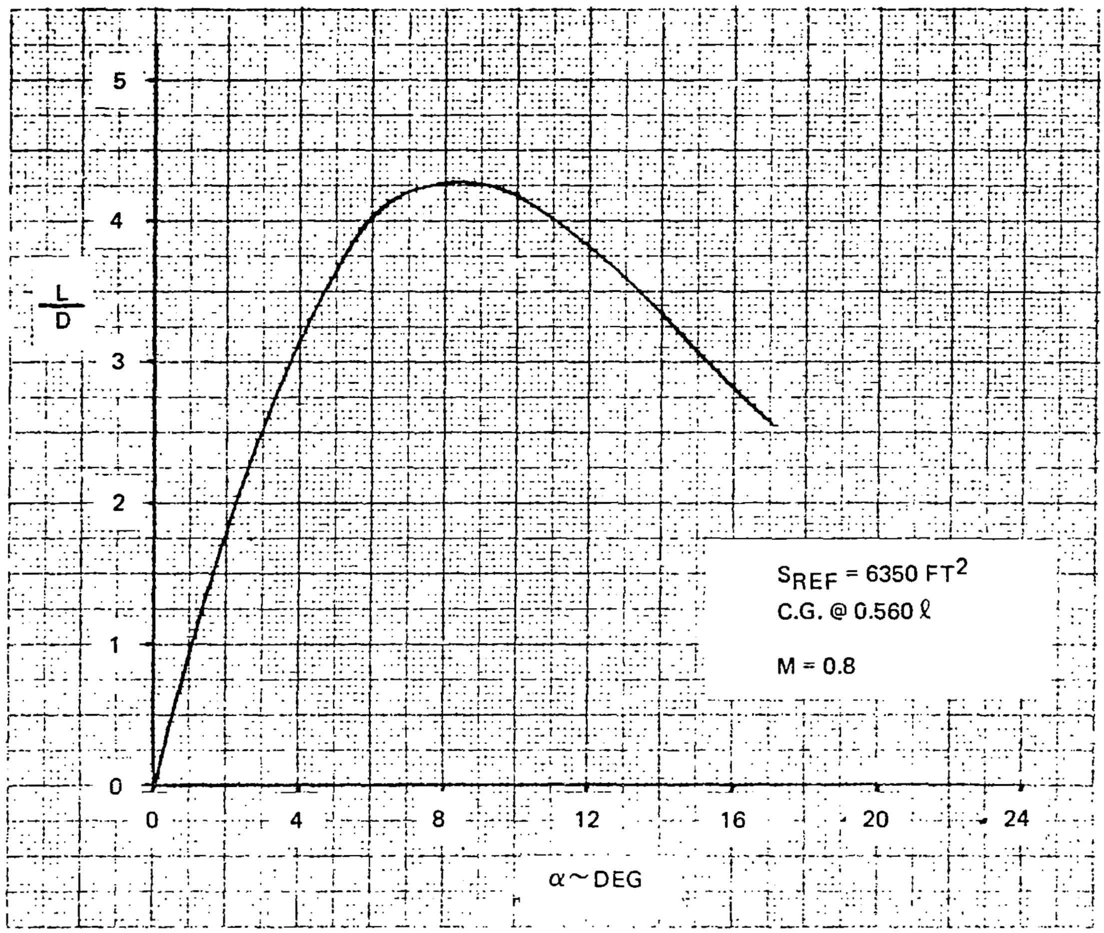



Figure 28. - Orbiter Subsonic Trimmed Lift-to-Drag Ratio

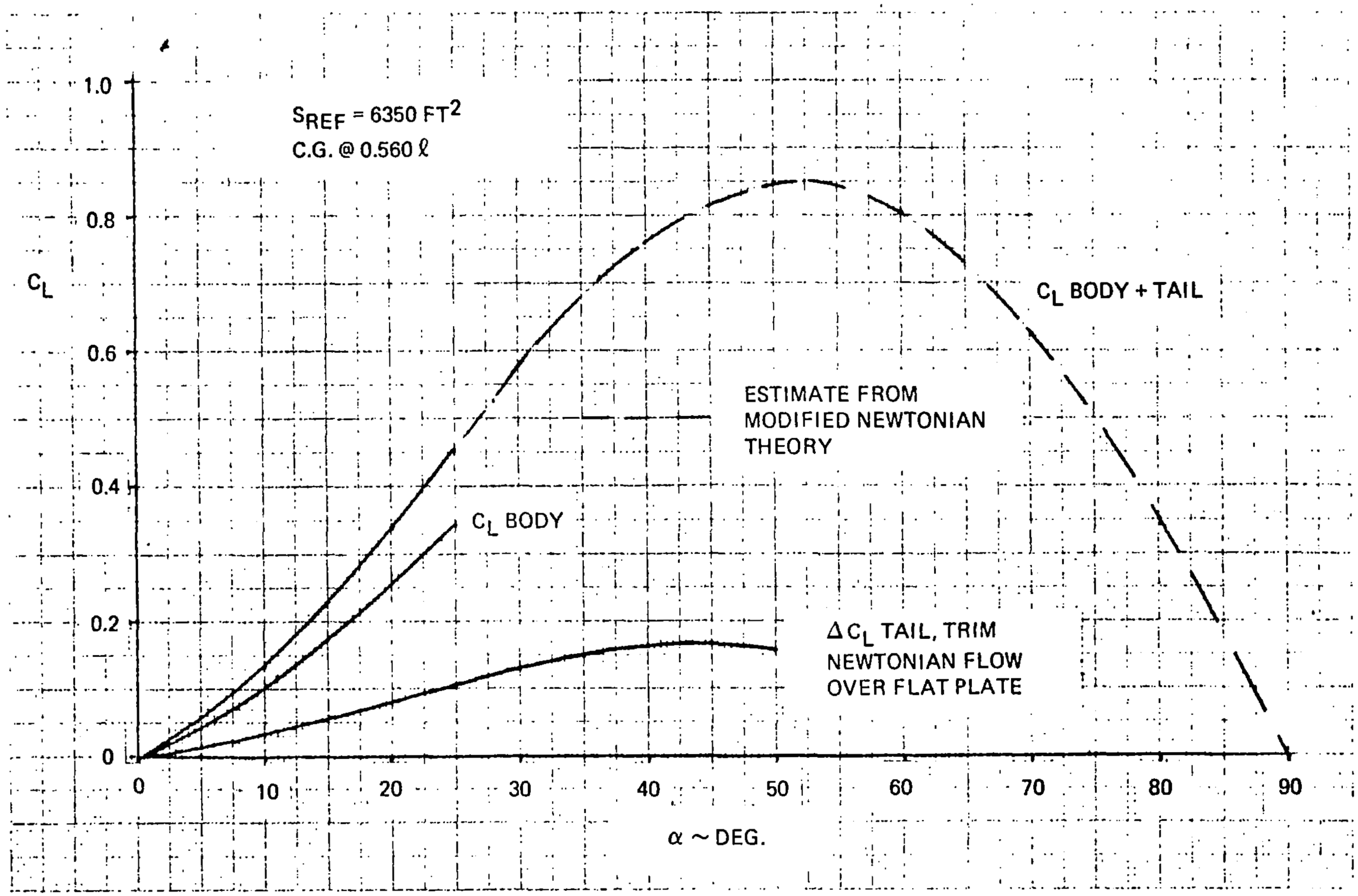



Figure 29.- Orbiter Hypersonic Trimmed Lift Coefficient

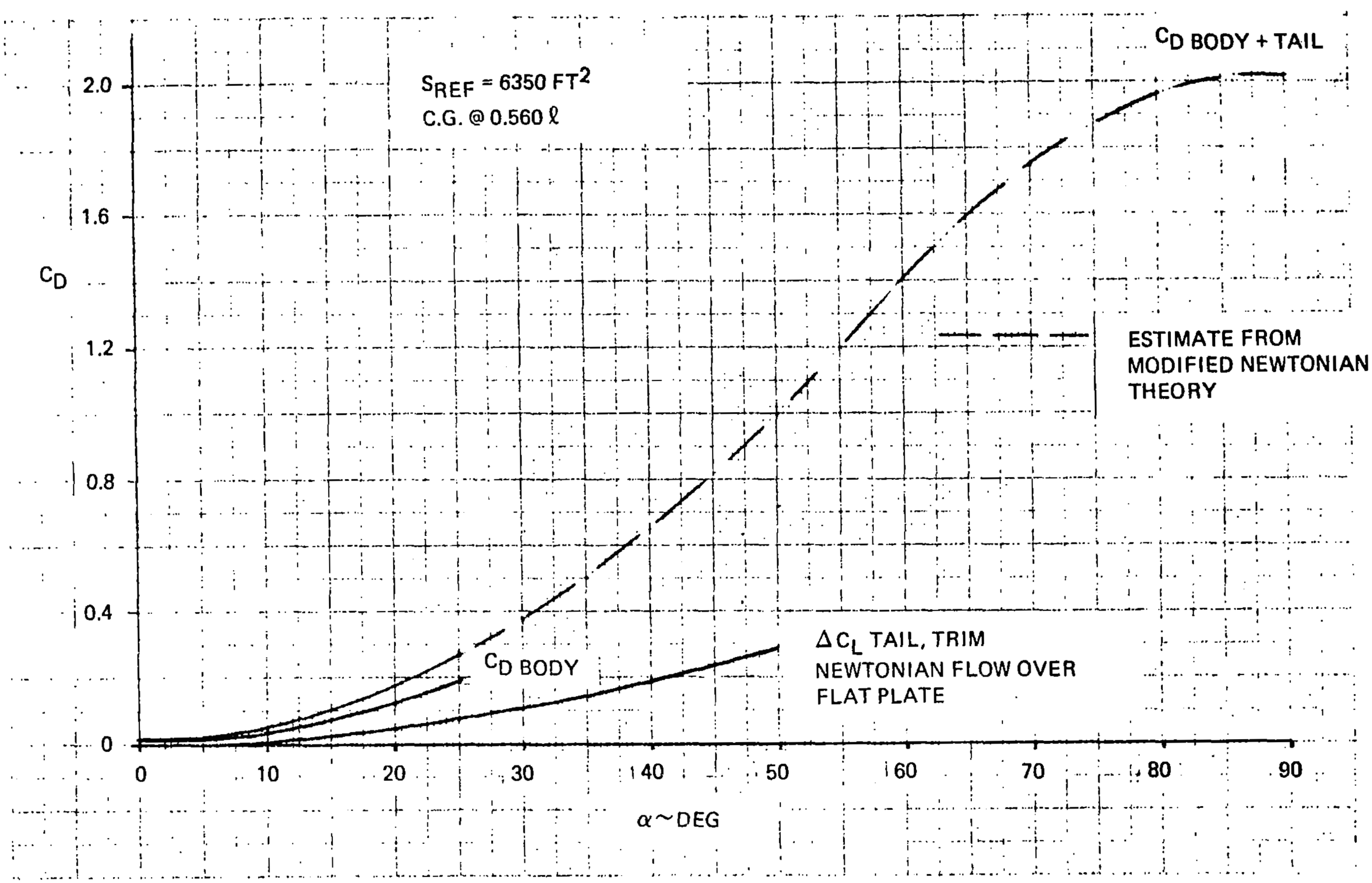



Figure 30. - Orbiter Hypersonic Trimmed Drag Coefficient

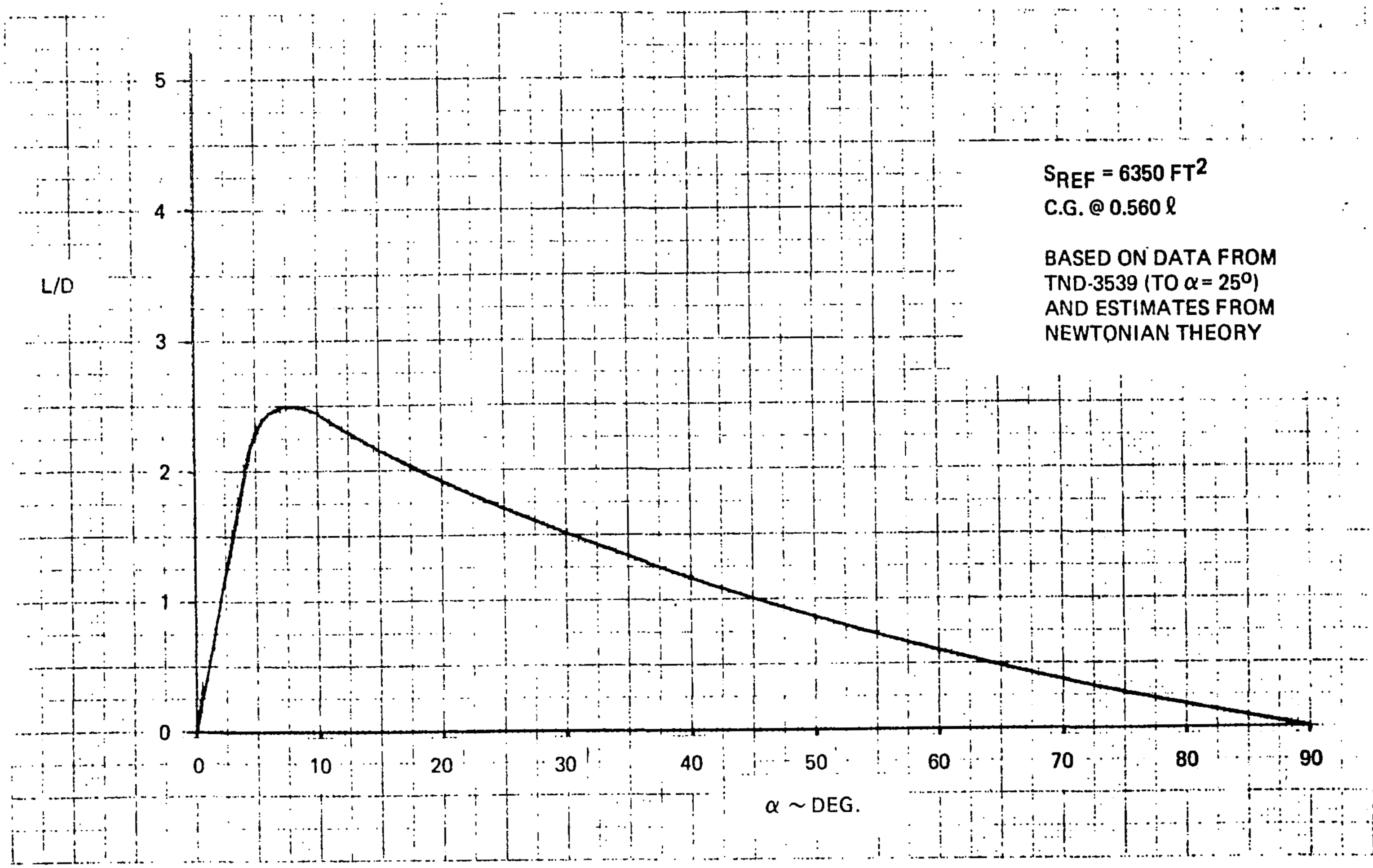



Figure 31. - Orbiter Hypersonic Trimmed Lift-to-Drag Ratio

to 11,400 square feet to approximate the effect of increased wing area for the booster capable of landing with the orbiter aboard.

Comparison of orbiter and booster hypersonic aerodynamic data and weight identifies a requirement for booster-orbiter rendezvous. For flight at equal lift-to-weight ratios, which is required for docking, the booster must trim to an angle-of-attack of about twice that for the orbiter, resulting in a considerably smaller lift-to-drag ratio. This causes the booster to lose velocity and altitude more rapidly than the orbiter. Boster thrust or orbiter drag control is required to maintain the same flight conditions. Maximum thrust required for the booster is approximately 200,000 pounds. Fuel usage would be about 27,000 pounds per minute. Orbiter speed brakes would require a total area of about 200 square feet. Another approach is to reshape the configurations for better hypersonic aerodynamic compatibility; however, some capability to vary thrust or drag would still be required to control relative positions of the two vehicles during terminal rendezvous and docking.

## 5.3 Airplane

The size and weight of the orbiter dictate that a large airplane be used to accomplish a spacecraft/aircraft rendezvous. The Lockheed C-5A was selected as the baseline airplane.

From Refs. 11 and 12 and other C-5A data the cruise parameters were found to be:

Mach Number = 0.8

Altitude = 36089 ft

Lift = Weight = 525,000 lb.

Drag = Thrust = 32,976 lb. (4 engines)

Ref. Area = 6200 ft.$^2$,

so that $C_{L_{CRUISE}} = 0.400$ and $C_{D_{CRUISE}} = 0.0250$. Lacking such explicit data for other flight conditions, the remainder of the aerodynamic coefficients were generated using the cruise $C_L$ and $C_D$ as initial inputs to the following simple approximations:

$$C_L = \frac{C_{L_{INCOMPRESS}}}{\sqrt{1 - M^2}}$$

Prandtl-Glauert Law for compressibility effects on lift coefficient; gives $C_L = C_L (M)$ at fixed $\alpha$.

$$C_L = C_{L_\alpha}(\alpha + 3.5^\circ)$$

For 3.5° wing incidence. $C_{L_\alpha}$ is based on NACA 0012 airfoil data from Ref. 13.

$$C_D = C_{D_o} + \frac{C_L 2}{\pi AR} \qquad C_{D_o} \text{ assumed independent of altitude and M.}$$

In view of the above assumptions, the aerodynamic coefficients will be inconsistent with those of the actual C-5A as deviations from the cruise point occur. However, the results should be at least representative of a large aircraft of the type required and, therefore, sufficient for the purpose of the present feasibility study.

Cruise Performance

To ensure satisfaction of the orbiter 1100 n.mi. lateral range requirement, the maximum radius capability of the aircraft on a rendezvous mission was estimated using Figures 22 and 23. For calculating performance, the mission was divided into pre- and post-rendezvous phases with these phases further divided according to whether the aircraft ultimately towed or docked the orbiter.

A docking or towing mode will affect pre-rendezvous performance primarily by fixing the amount of fuel the aircraft can accommodate without exceeding its in-flight weight limit. To always remain under 728,000 lbs and still carry the maximum amount of fuel is not a problem with the towing concept, but the eventual acquisition of orbiter weight ($\approx$200,000 lbs.) in the docking case means carrying about 75,000 lbs. less fuel initially. Another important consideration in computing the performance is the weight of orbiter support equipment. A summary of the significant pre-rendezvous weight estimates follows:

| | Towing Concept | Docking Concept |
|---|---|---|
| Operating Wt. of C-5A | 323,904 lbs | 323,904 lbs |
| Estimated Orbiter Support Equip. | 20,000 | 34,100 |
| Total Fuel | 318,500 | 243,500 |
| Take-Off Weight | 662,404 lbs | 601,504 lbs |
| Reserve Fuel | 34,900 lb | 31,200 lb. |

For each rendezvous concept, an allowance was made for climb-to-cruise range and fuel consumption. The outbound cruise phase, itself, was determined as follows:

$V$ = 440 knots, true air speed

$\dot{W}$ = Installed Cruise Fuel Flow Rate, $\frac{\text{lbs}}{\text{hr}}$

$SR = \frac{V}{\dot{W}}$ = Specific Range, $\frac{\text{n.mi.}}{\text{lb. fuel}}$

$R_{CRUISE} = (\overline{SR})(W_F)$ = Cruise Range, n.mi. $\overline{SR}$ is average of specific ranges at beginning and end of cruise; $W_F$ is wt. of fuel consumed during cruise.

Total pre-rendezvous range is then just the sum of the climb and cruise contributions and will equal the post-rendezvous result with a correct split of the available fuel.

Having achieved rendezvous (performance during rendezvous maneuver neglected) the airplane experiences a significant weight and drag increase. Return flight performance was based on estimated aerodynamic data (from above equations) and representative engine data. While not strictly applicable to the C-5A, the data may be expected to yield representative performance estimates.

In the case of a rendezvous and tow mission, aircraft performance was based on the flight condition which enabled the orbiter to be towed at (L/D) max with a minimum cable tension (Figures 22 and 23), namely,

V = 246 knots equivalent airspeed

= 265 knots,true airspeed at 5000 ft.

$\alpha$ Orbiter (Tow) = 8°, (L/D) max condition

$\theta$ = 15°, tow cable angle

t = 45,000 lbs, tension in cable under above conditions.

(NOTE: Figures 22 and 23 assume a straight cable and neglect cable lift and drag.)

The effective aerodynamic forces were estimated as follows:

$$L = W_{A/C_{alone}} + t \sin\theta$$

Lift required to maintain level flight.

$$\approx 604{,}000 \text{ lbs.}$$

$$C_L = \frac{L}{qS}$$

$$C_{D_{A/C\ alone}} = C_{D_o} + \frac{C_L{}^2}{\pi AR}$$

$$D = T = C_{D_{A/C\ alone}}\, qS + t \cos\theta$$

Total drag experienced by aircraft = Thrust required for cruise.

$$= 79{,}500 \text{ lb}$$

A fuel flow rate commensurate with the required thrust was estimated, and cruise range was computed as shown previously. To develop and maintain the required thrust, however, it was found necessary to cruise back entirely at the assumed rendezvous completion altitude of 5000 ft. A 20-80% out-back available fuel split yielded a maximum towing mission radius of about 1370 n.mi.

A low cruise-back altitude (5000 feet or less) is required to develop sufficient thrust to overcome the large drag increment of the orbiter in the towed condition with four-engine operation. Three-engine operation provides insufficient thrust. Uprated engines, or an additional engine, would be required to provide satisfactory performance with an engine failure.

Booster recovery by airplane towing was also investigated. Total drag is about 105,000 pounds at 155 knots EAS. It was found that four engine operation provides inadequate thrust, even for very low altitude conditions. Two additional engines would be required to provide a low altitude cruise capability with one engine inoperative.

In the case of a rendezvous and dock mission, aircraft performance was based on the assumption of a 50% increase in airplane drag with the orbiter aboard. Initial speed and altitude for cruise back was assumed identical to those for the towed orbiter. The aerodynamic forces effective under these conditions were estimated as follows:

$$L = W = W_{A/C\ alone} + W_{orbiter} = 726{,}600\ \text{lbs.}$$ Lift required to maintain level flight

$$C_L = \frac{L}{qS}$$

$$C_{D_{A/C\ alone}} = C_{D_o} + \frac{C_L^2}{\pi AR}$$

$$D = T = 1.5\, C_{D_{A/C\ alone}}\, qS = 62{,}500\ \text{lbs.}$$ Total drag experienced by aircraft = Thrust required

Since cruising at 20,000 ft. instead of 5,000 ft. gave improved fuel flow rates without jeopardizing required thrust levels, return cruise performance for the aircraft-docked orbiter combination was based on a 20,000 ft. altitude at 265 knots (true airspeed). A one-half hour climb to curise altitude was included in the calculations. A 30-70% out-back available fuel split yielded a maximum docking mission radius of about 1430 n.mi.

Glide Requirements

At the start of the docking or towing maneuver, the aircraft must have the same flight conditions as the orbiter.

Glide Path Angle = 13.2°

Speed = 372 knots (true airspeed)

= 0.65 M

Altitude = 36,100 ft.

Weight of Aircraft = 526,600 lb.

Four Engines assumed operating at one-half throttle ( $\approx$ 16,500 lbs. thrust) gives aircraft potential to accelerate or decelerate as required.

A large drag force of about 137,000 pounds is necessary to maintain the desired glide path angle. Since the basic aircraft contributes only about 28,600 pounds toward the total required, a large deceleration device is needed to generate additional drag. Three such devices were considered: trailing-edge dive brakes, a drag parachute, and wing flaps modified to deflect upward.

The dive brake and drag chute configurations were assumed deployable without affecting the basic aircraft lift characteristics. The size of brakes or chute, therefore, could be approximated in a straight-forward manner to supply a drag increment of 137,000-28,600 = 108,400 pounds. Required dimensions are 22.5' x 6.2'/panel for dive brakes deflected 60° or 47.6' x 6.2'/panel for brakes deflected 30° (Ref. 14), and 42.4' diameter for a ring-slot type parachute (Ref. 15).

The application of wing flaps as decelerators was analyzed separately from that of dive brakes or drag chute because deflected wing flaps affect aircraft lift and drag simultaneously. In their capacity as decelerators, the flaps are presumed deflected trailing edge up. This action forces the aircraft to fly at a higher angle of attack in order to develop the lift required for maintaining the glide path angle. Coincidentally, the drag-required during glide will result from (1) the aircraft at the higher $\alpha$ and (2) the deflected flaps.

The change in lift with flap deflection was based on NACA 0012 airfoil data (Ref. 13). (The C-5A uses a modified NACA 0012 airfoil.) A linear relationship was assumed between lift coefficient and flap deflection at a fixed angle of attack, and upward flap deflections were assumed to produce an equal but opposite change in lift from equivalent downward deflections. The above assumptions should be valid over the moderate ranges of angle-of-attack and flap deflection involved.

The drag variation was estimated using a theoretical expression from Ref. 13 which gives flap normal force coefficient as a function of wing section lift coefficient and flap deflection. A flap drag component, determined from

the normal force coefficient, was then added to the no-flap drag to get the total drag estimate.

The preceding lift and drag yield reasonable combinations of aircraft angle-of-attack and flap configurations. The following are typical combinations to provide the lift and drag necessary for maintaining the desired glide path angle:

(1) Two 73' x 6.8' flaps
Deflection $\approx 21^\circ$ (trailing-edge up);
Aircraft angle of attack = $7^\circ$

(2) Two 50' x 6.8' flaps
Deflection $\approx 31^\circ$ (trailing-edge up);
Aircraft angle of attack = $7^\circ$

## 5.4 Lateral Range

The basic hypersonic lateral range capability of the orbiter is nearly 2000 nautical miles. This may be determined from any one of a number of sources providing parametric data of this type, such as Reference 16. Some loss in maximum lateral range occurs due to maneuvers for thermal control, guidance, rendezvous, and docking. These effects should be small, with the possible exception of thermal control. Assuming that the loss can be held to a minor fraction of the basic capability, it is thought that the orbiter alone can exceed the Space Shuttle Phase B lateral range requirement of 1100 nautical miles. The orbiter-airplane rendezvous results in a large additional lateral range capability due to the airplane's radius-of-action of 1400 nautical miles. Some component of the booster's subsonic cruise range could also be added to the orbiter's capability for the orbiter-booster rendezvous. The Phase B boosters are designed to cruise about 400 nautical miles. It is concluded that lateral range capability is good for both cases, especially for orbiter-airplane rendezvous, which exceeds the requirement by at least a factor of two.

## 6.0 TRAJECTORY DATA AND RELATIVE MOTION

Orbiter and booster reentry-glide trajectories were calculated using a digital computer routine employing three degree-of-freedom (point mass) equations and assuming no earth rotation or thrust forces. Vehicle aerodynamic definition consisted of vehicle drag coefficient as a function of angle of-attack and Mach number, and variation of vehicle normal force coefficient with angle-of-attack and normal force coefficient at zero angle-of-attack, both as functions of Mach number. Also included was desired angle-of-attack as a function of Mach number.

The computer routine includes a capability to vary angle-of-attack to damp altitude oscillations; however, optimum control gains were not determined for this study. The gain was arbitrarily set at a relatively low value, resulting in small variation of angle-of-attack from reference input values. As a result, trajectory and relative motion data include effects of altitude oscillations that may be somewhat exaggerated compared to data that would be obtained for vehicles with effective damping control. This does not appear to affect the more significant results of the trajectory studies.

Orbiter velocity from hypersonic to subsonic speeds is shown in Figure 32 as a function of range to rendezvous and angle-of-attack (for the subsonic rendezvous case). Angles-of-attack of 17° and 50° correspond to near maximum and minimum usable hypersonic lift-to-drag ratios. The data for both cases are based on transition to 10° angle-of-attack for favorable subsonic glide and recovery characteristics. Figure 32 shows that considerable capability is available to correct position errors during the hypersonic glide.

Typical velocity versus range for an orbiter-booster rendezvous is shown as Figure 33 from booster lift-off to beyond rendezvous. These data were prepared by combining reentry-glide computations with booster launch characteristics based on North American Phase B Space Shuttle studies. At booster lift-off, the orbiter is approximately 225 nautical miles uprange from the launch site and at a velocity of about 13,000 feet per second. At booster apogee, the orbiter is approximately 50 nautical miles downrange from the booster and at a velocity of about 9,000 feet per second. Rendezvous occurs at a velocity of 5000 feet per second and about 500 nautical miles downrange from the launch site. Corresponding altitude versus velocity data are shown in Figure 34. Apogee for the booster is established by launch of another orbiter. Due to apogee being well above equilibrium glide altitude, the first booster overshoot of orbiter flight altitude shown in Figure 34 can not be avoided. It appears that rendezvous at speeds below 6,000 feet per second can be accomplished by proper control of angle-of-attack. Rendezvous at higher speeds would be very difficult unless the booster launch trajectory were reshaped.

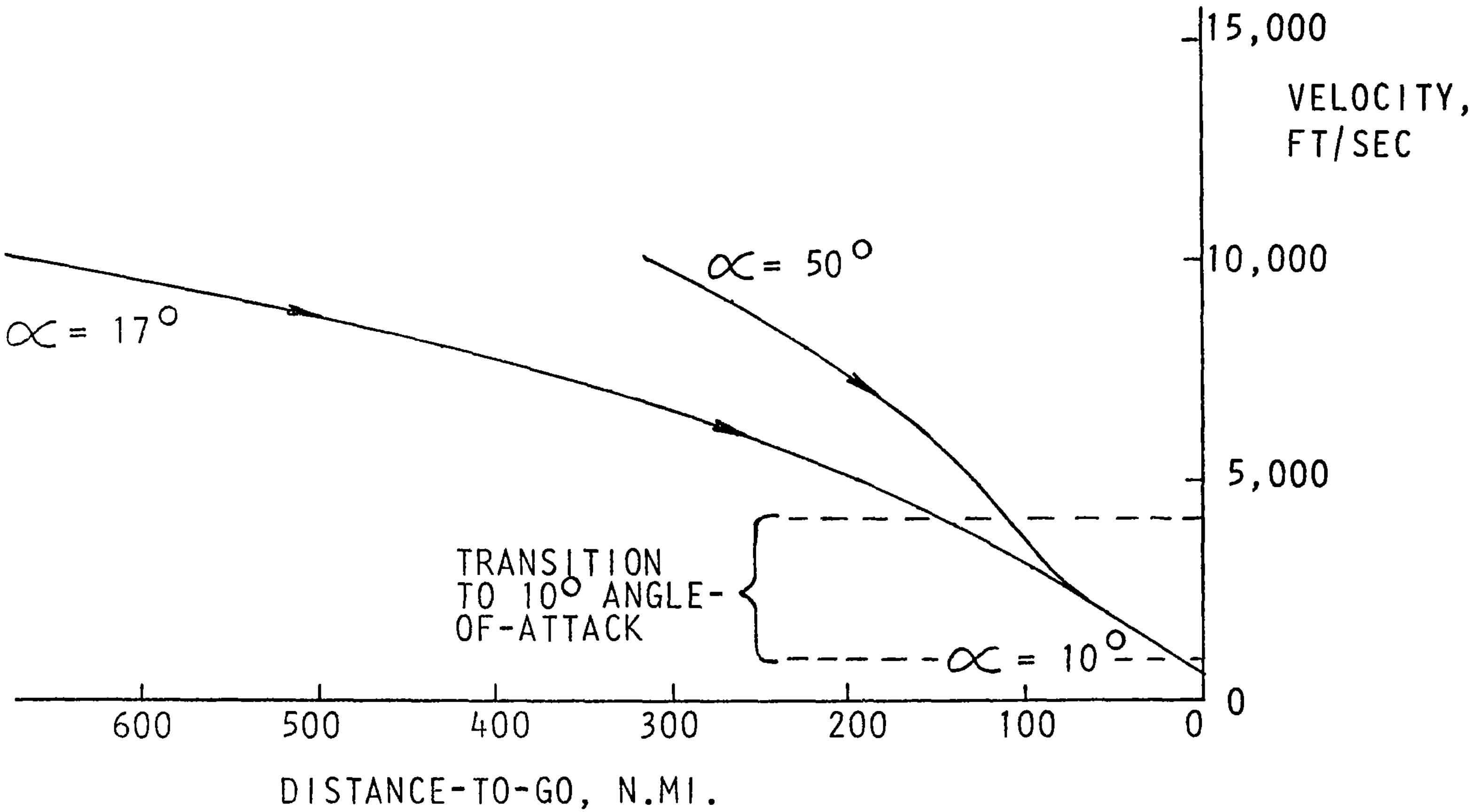



Figure 32. - Orbiter Maneuver Capability

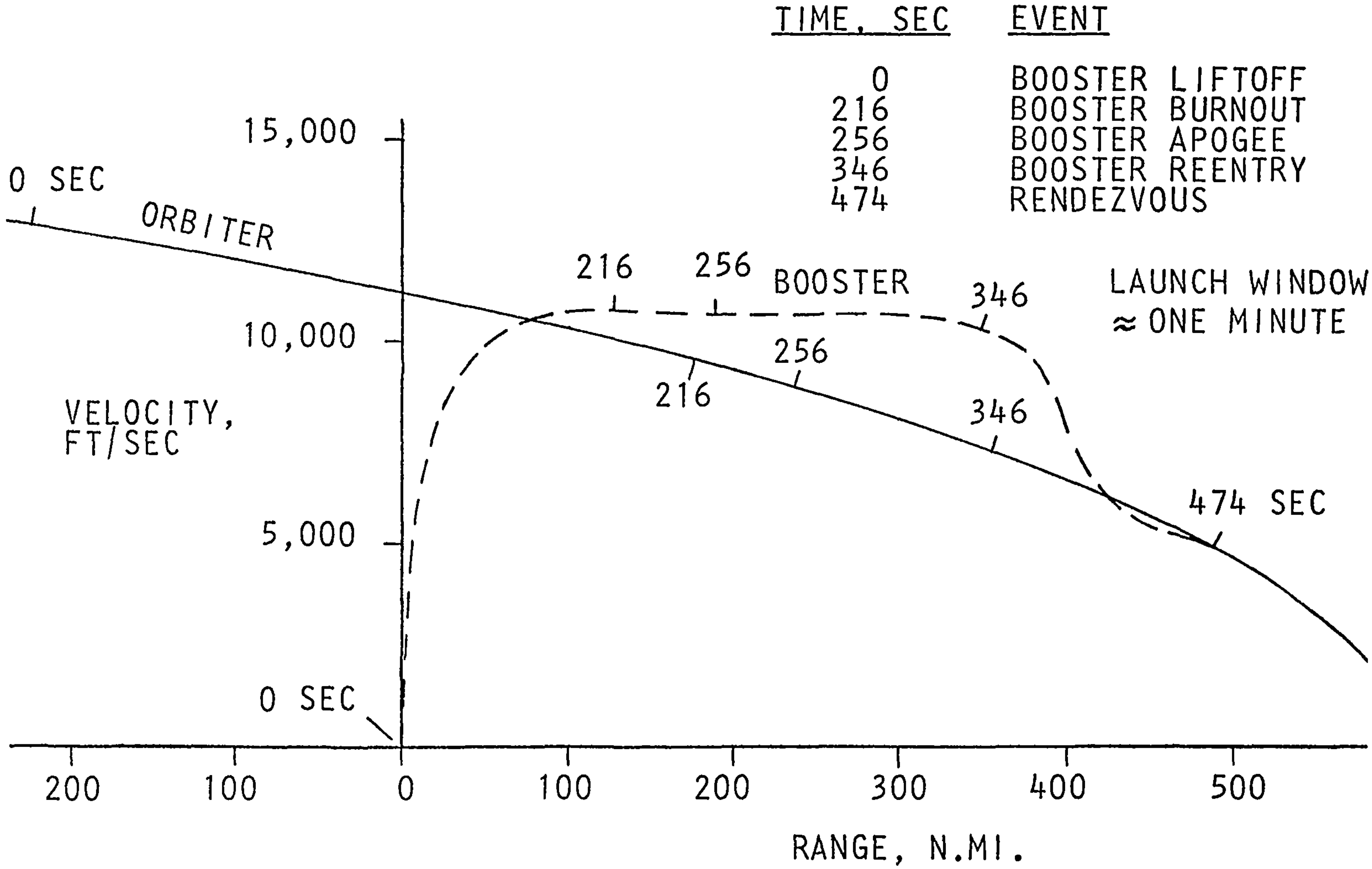



Figure 33. - Typical Orbiter-Booster Rendezvous

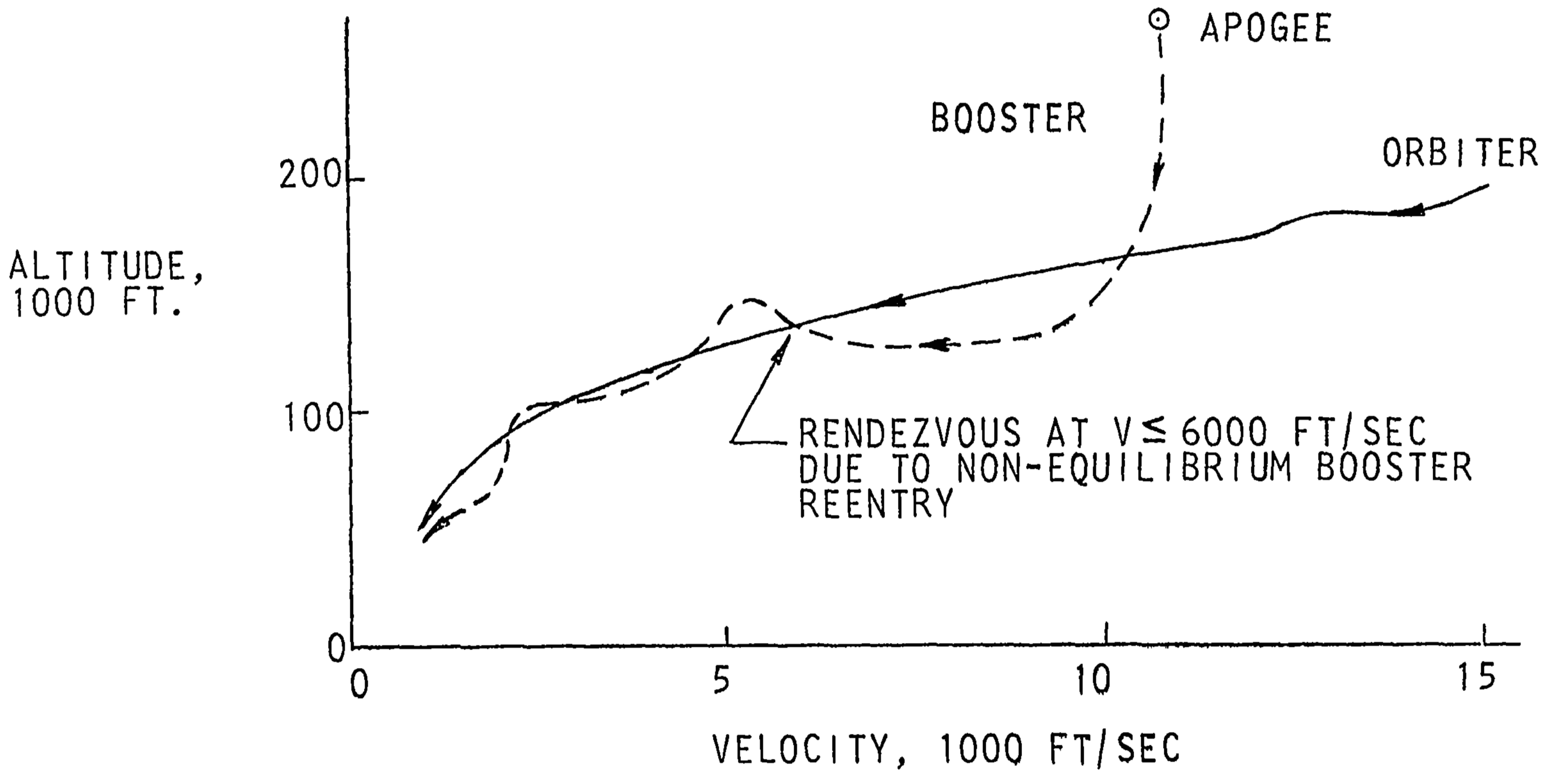



Figure 34. - Orbiter and Booster Reentry

Space Shuttle studies generally assume a hypersonic turn by the booster after reentry, resulting in a subsonic return cruise of about 400 nautical miles. This turn would probably not be made during a hypersonic rendezvous; therefore, the return cruise would have to be somewhat longer (see Figure 33) or a landing site would have to be available downrange from the launch site.

Airplane trajectories were also calculated using the digital computer routine. Simulation was similar to the booster and orbiter except a thrust force was used to offset the vehicle drag. The airplane trajectory remains at nearly a constant altitude and velocity by varying the angle-of-attack.

Airplane maneuver capability to correct position errors late in the rendezvous phase is indicated by Figure 35. These data were computed in earlier studies at the NASA Langley Research Center. It was found that a range control capability of 19,000 feet exists during transition from a nominal airplane cruise condition to orbiter glide conditions in the final two minutes of rendezvous.

Relative motion during rendezvous was computed without simulation of a rendezvous guidance system. The vehicles were placed at the same point in space and the equations of motion were integrated backwards for some 500 seconds. The conditions at 500 seconds were then used as initial conditions for the rendezvous trajectories. In addition to calculating earth relative conditions of the two vehicles, conditions of the booster and aircraft relative to the orbiter were also obtained from another digital computer routine. The coordinate system used to define the relative conditions is shown in Figure 36. This coordinate system's origin remains with the orbiter with the X-axis along the local horizontal and in the direction of motion. The Z-axis is along the local vertical so the X-Z plane coincides with the orbit plane. For simplicity, the Y-axis components of displacement and velocity, which are the out-of-plane components, were neglected, limiting these studies to coplaner conditions.

## 6.1 Orbiter-Airplane Rendezvous

Relative motion of the orbiter and airplane during the final 500 seconds prior to rendezvous is shown in Figures 37 and 38. As shown, the orbiter initially approaches the airplane from the rear and above. Relative motion during the final 200 seconds is generally in the vertical direction, as shown by Figure 38. The "circling" effect shown during the final 90 seconds is due to altitude oscillations and lack of rendezvous guidance in the relative motion computations. Airplane maneuvers can be employed during the terminal phase of the rendezvous to maintain an efficient continuation of the vertical motion, as indicated by the dashed line.

Time available to perform a docking is illustrated in Figure 39. Relative motion data and rendezvous guidance studies (Section 7.0) indicate

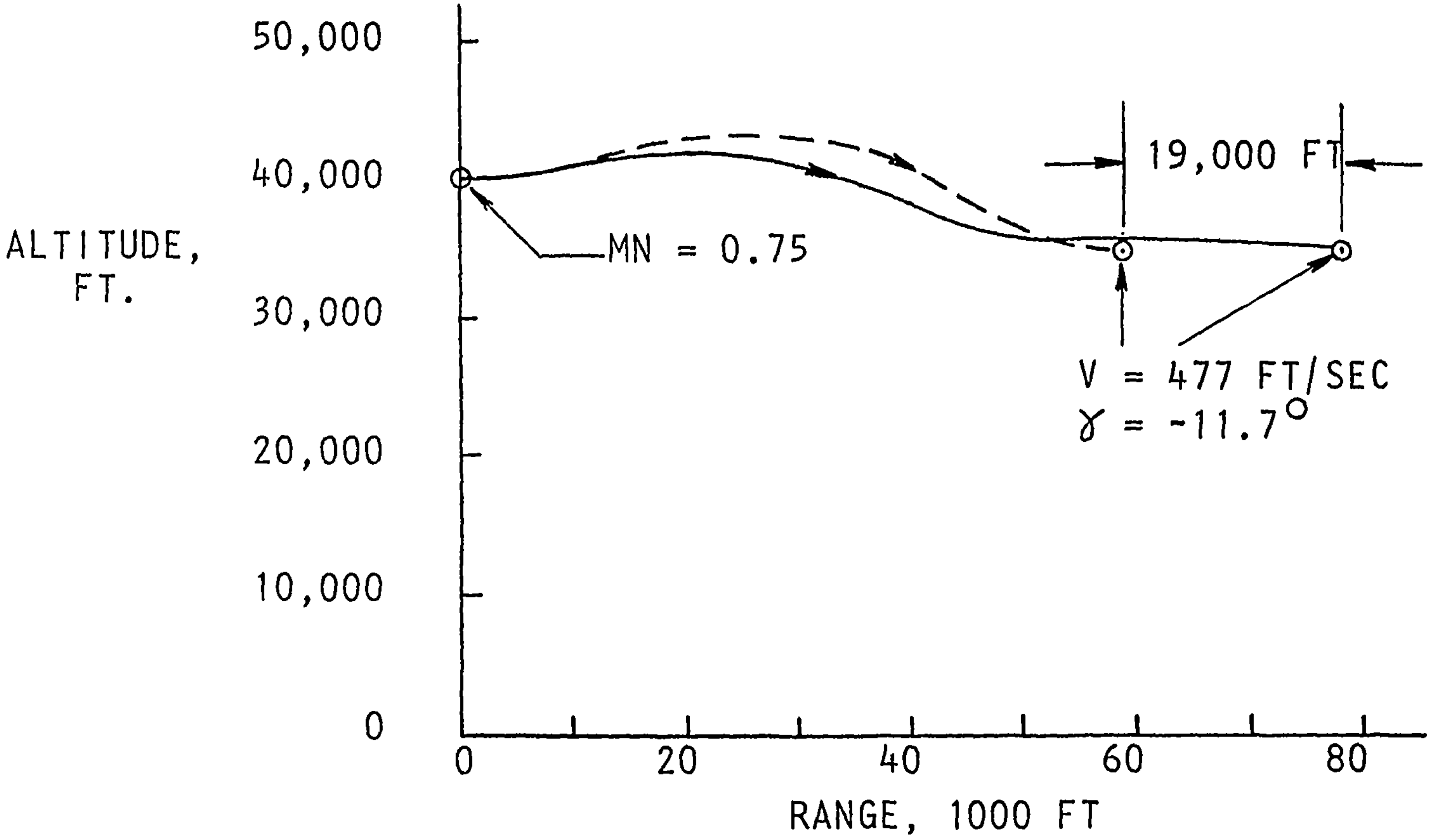



Figure 35. - Airplane Maneuver Capability

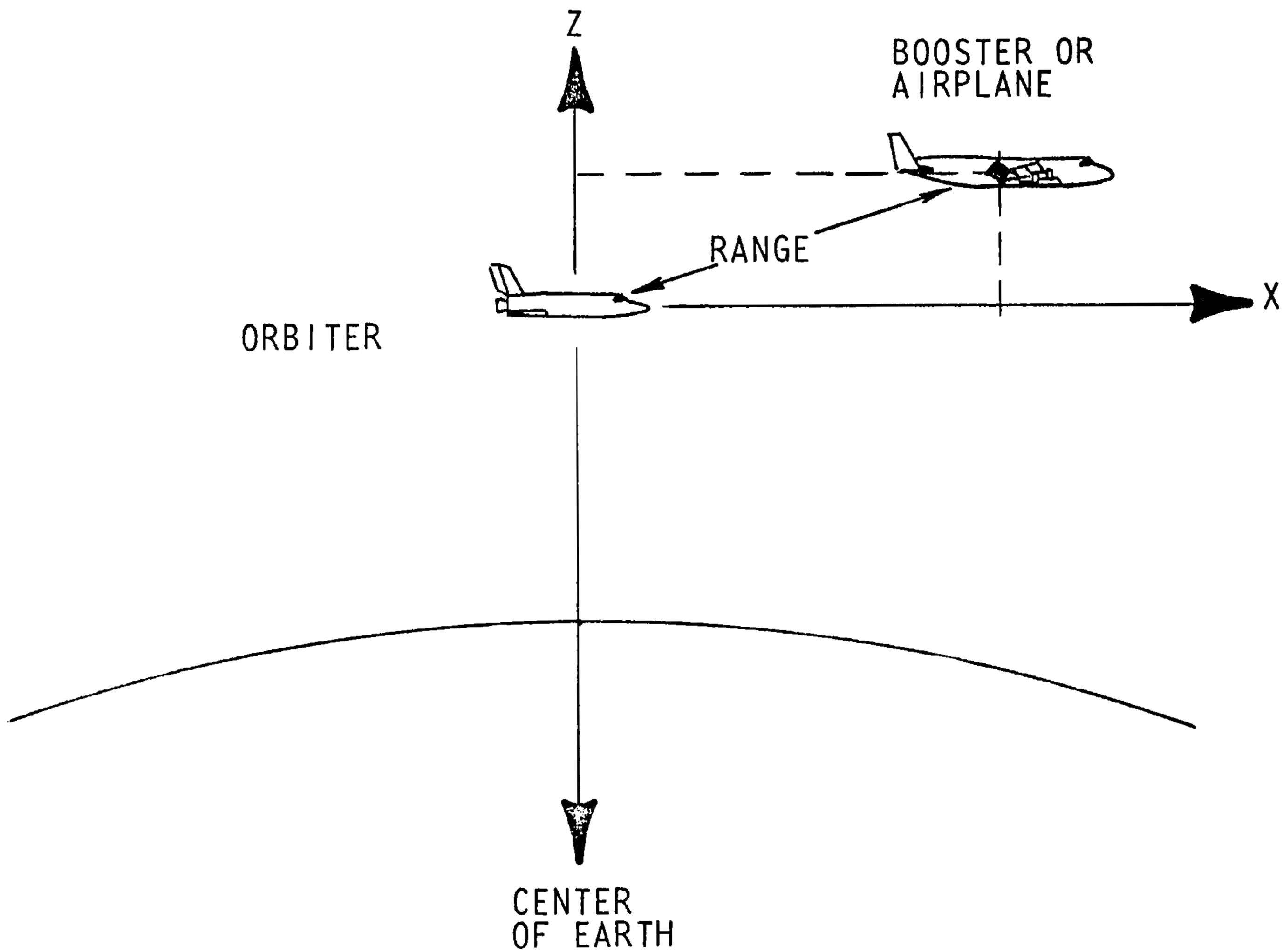



Figure 36. - Axis System

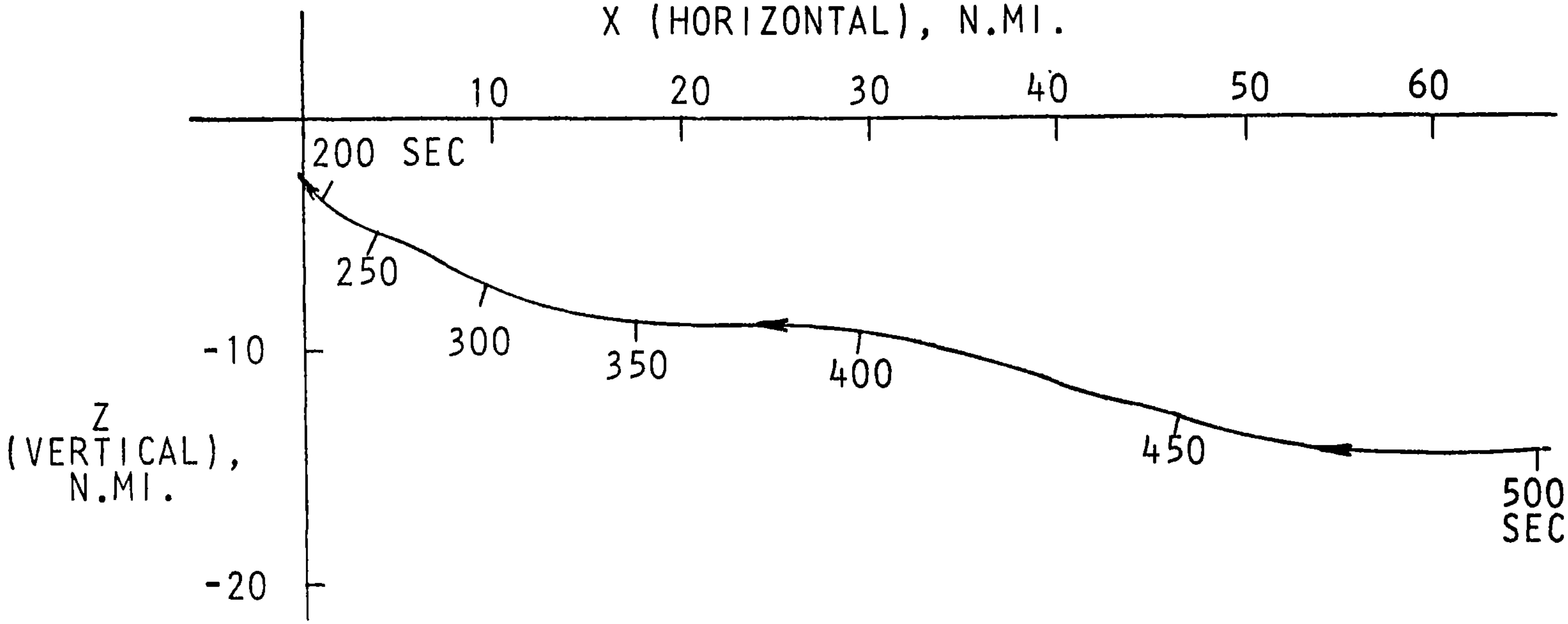



Figure 37. - Airplane Motion Relative to Orbiter (500 to 200 Seconds-to-Go)

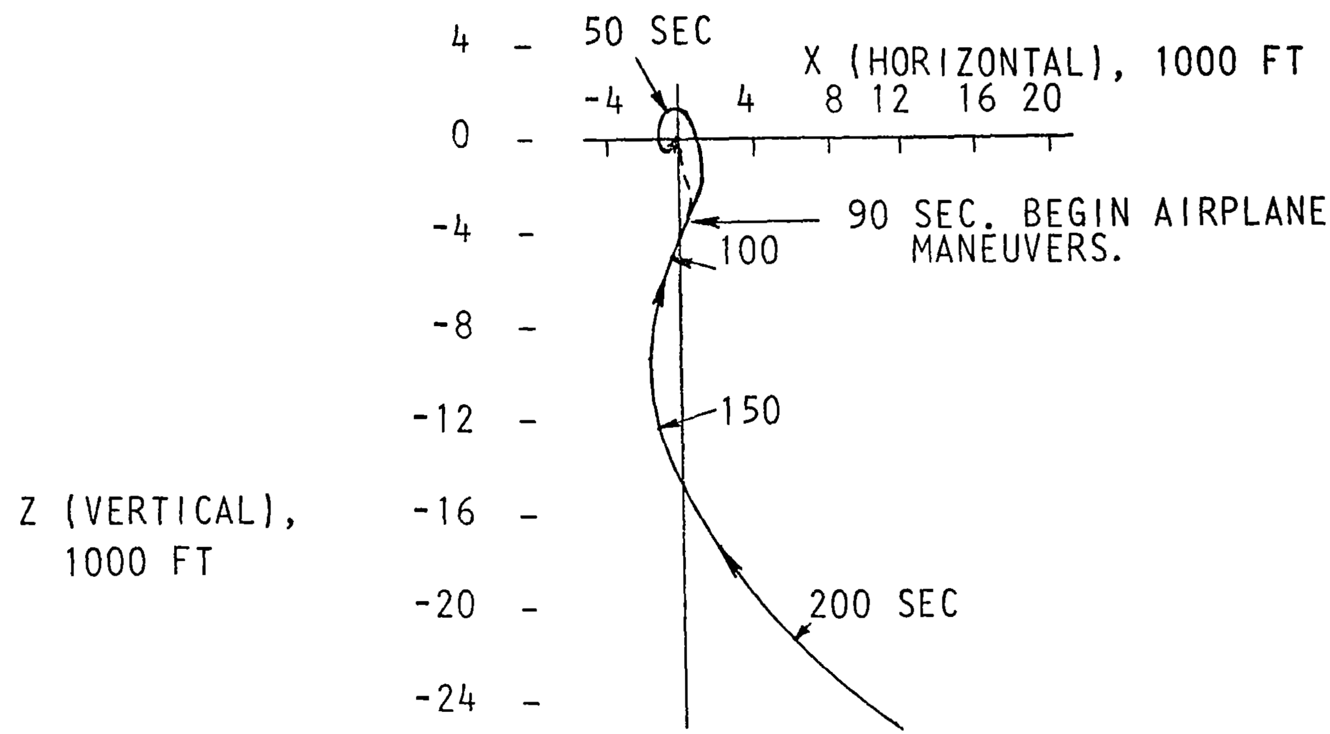



Figure 38. - Airplane Motion Relative to Orbiter (Final 200 Seconds)

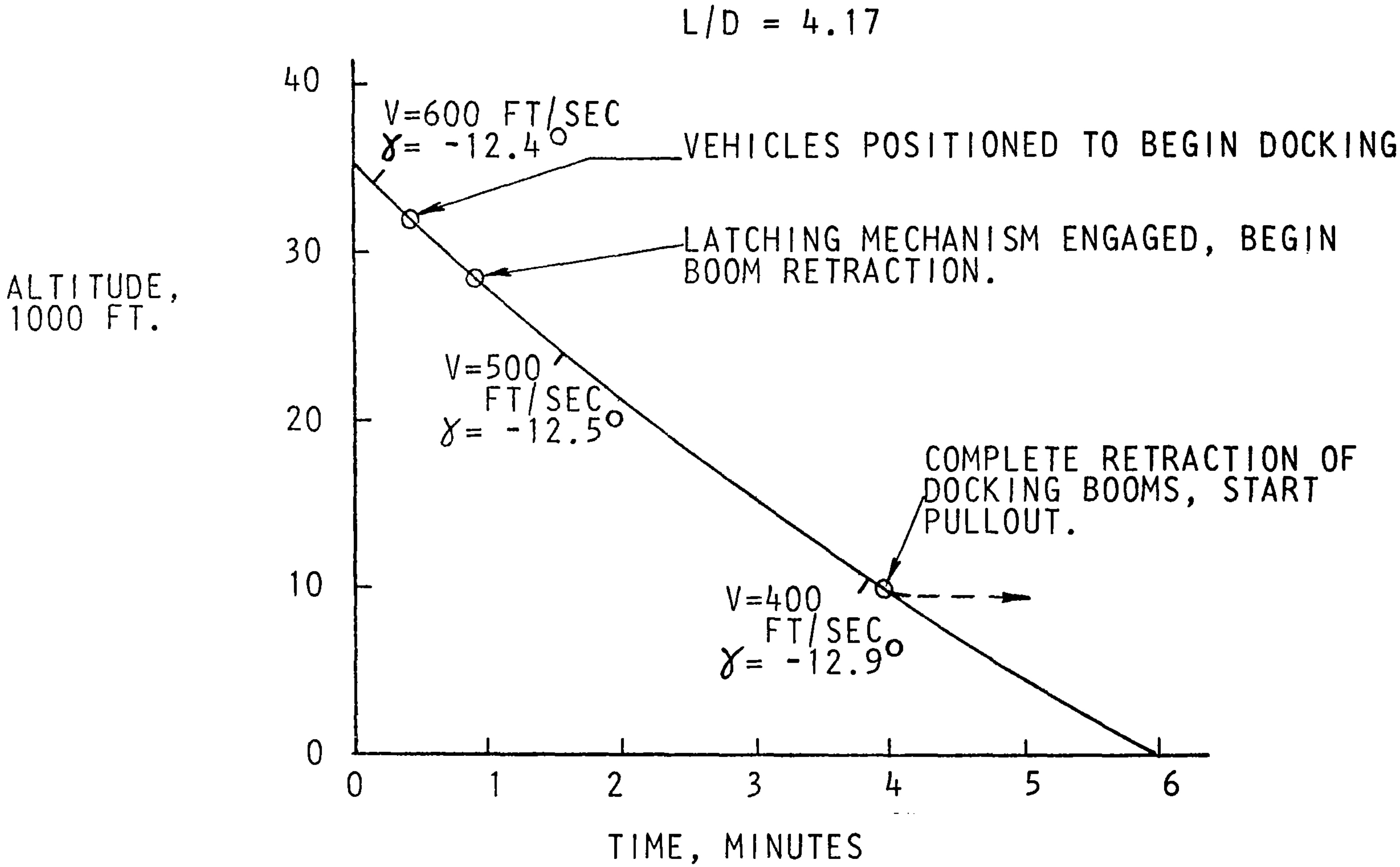



Figure 39. - Orbiter Subsonic Glide

that the airplane and orbiter vehicles can be positioned to begin docking at an altitude of 32,000 feet. Thirty seconds are allowed for engaging the latching mechanism on the main telescoping boom. This is reasonable based on modern air-to-air refueling operations. An additional three minutes are available before the vehicles descend to an altitude of 10,000 feet, which is considered a safe altitude for completion of docking and pulling out of the glide. Pullout prior to completion of the docking maneuver would impose excessive loads on the boom due to aerodynamic drag on the orbiter. The telescoping boom can be fully retracted within two minutes; therefore, the available time is adequate. The additional altitude loss during pullout is about 500 feet. Additional time is available, if required, by reducing the minimum pullout altitude.

The towing mode of orbiter recovery is less sensitive to timing problems, because pullout can be initiated as soon as the tow cable is engaged.

## 6.2 Orbiter-Booster Rendezvous

Range versus range-rate data for rendezvous at speeds of 4000 and 8000 ft/sec are shown in Figure 40. It is thought that the oscillations seen in these curves would be eliminated or greatly reduced for vehicles utilizing efficient damping of altitude rate. Note that the curves are very similar for the two cases. This indicates that a rendezvous guidance procedure could be established that is independent of earth-relative velocity. The procedure would probably involve measuring or computing range and range-rate and maneuvering one or both vehicles to maintain a range range-rate schedule close to a pre-determined nominal curve.

The booster is gliding at a higher speed and a smaller lift-to-drag ratio than the orbiter during the rendezvous flights shown in Figure 40. Therefore, the booster is continuously approaching from the rear of the orbiter. The relative altitude, however, is much less consistent, as shown in Figure 41 for the case of rendezvous at 4000 ft/sec. This plot of relative altitude versus relative altitude rate shows that the variation is well behaved only during the final minute of rendezvous. Although this is probably exaggerated by the poor damping of altitude rate, it indicates that relative altitude is a poor guidance parameter until late in the rendezvous. Early attempts to control relative altitude would probably have an adverse effect on range control. The use of thrust or drag devices in addition to angle-of-attack and bank angle control during the final phase of rendezvous would enable control of all components of relative position and relative velocity.

Figure 33 shows that booster launch occurs during the orbiter hypersonic glide. Therefore, there must be some constraint on the launch time in order to rendezvous. A study was made to estimate this booster launch window restriction.

Booster launch time can be delayed if its flight time to rendezvous

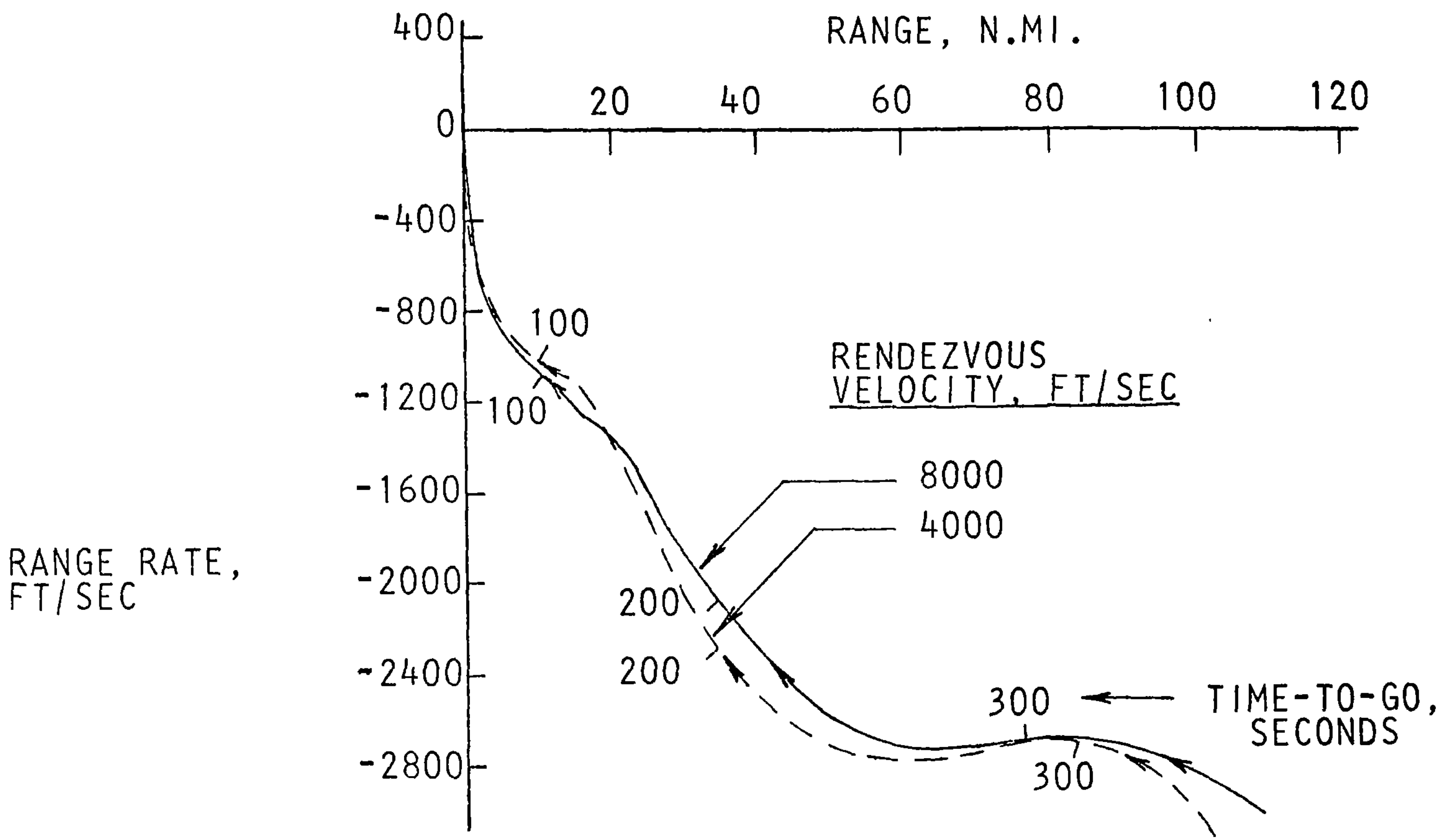



Figure 40. - Orbiter-Booster Relative Motion (Range)

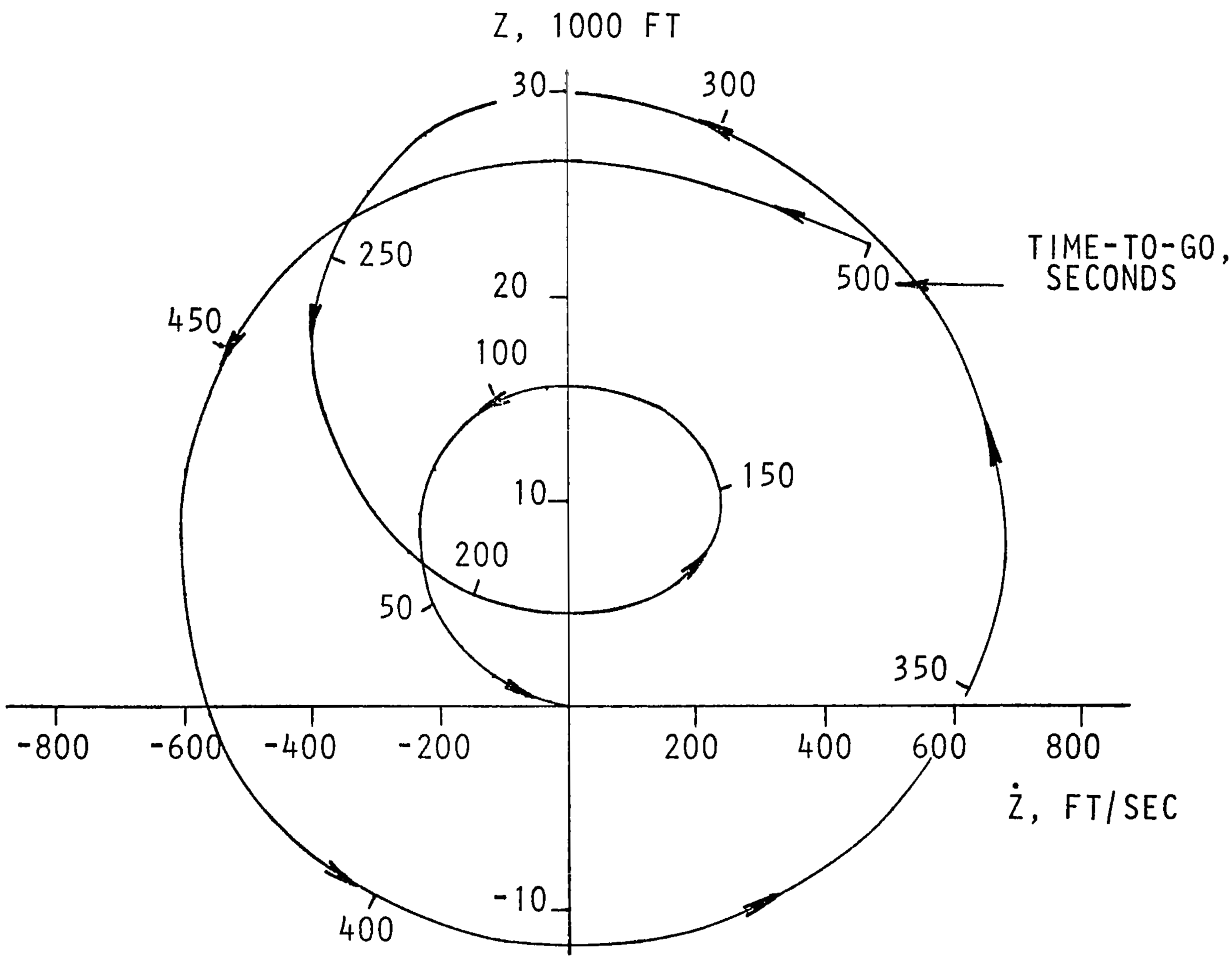



Figure 41. - Orbiter-Booster Relative Motion (Altitude)

is decreased and/or if the orbiter flight time to rendezvous is increased. The orbiter cannot delay re-entering since it is in the re-entry phase at the time of nominal booster lift-off.

The following two cases were considered for a rendezvous at 5000 fps, (1) booster flight was held fixed and orbiter maneuvers were used to increase the orbiter flight time to rendezvous and (2) orbiter flight was held fixed and booster maneuvers were used to decrease the booster flight time to rendezvous. In both cases, velocity and range at rendezvous were held constant. The nominal rendezvous is based on flight at average lift-to-drag ratios.

These data were calculated using equilibrium glide equations.

Case (1) - Orbiter Maneuvers - Orbiter maneuvers are initiated at nominal booster lift-off time if the booster cannot lift-off. Analysis showed that flight time to rendezvous is maximized by flying at minimum L/D (0.85) for 95 seconds and then maximum L/D (2.5) to rendezvous. This procedure increases the orbiter flight time to rendezvous by 48 seconds over the nominal time based on an average L/D (1.68). Thus the maximum booster launch delay time, based on the above procedure, is 48 seconds. If the orbiter uses the above defined maneuvers, the booster must be launched 48 seconds past the nominal time. However, the orbiter trajectory can be adjusted so that the orbiter will reach the rendezvous point anytime from nominal time to 48 seconds beyond nominal time.

Case (2) - Booster Maneuvers - As the booster begins its equilibrium glide at apogee, it rapidly loses altitude with very little loss in velocity since it is so far off its equilibrium glide path. Prior to reaching its equilibrium altitude, its flight path is insensitive to vehicle orientation. Therefore, booster maneuvers were initiated at 90 seconds beyond apogee when it is near its equilibrium altitude. Similarly to the orbiter maneuvers, booster flight time is minimized by flying at maximum L/D (1.7) for 75 seconds and then minimum L/D (0.7) to rendezvous. This procedure reduces booster flight time to rendezvous by 19 seconds over the nominal flight time based on an average L/D (1.2). In this case, the maximum booster launch delay time is 19 seconds. Booster flight time can be adjusted to anytime between nominal time and 19 seconds beyond nominal time. It should be noted that the booster has about 130 seconds to maneuver while the orbiter has about 475 seconds.

Assuming that the capabilities of these two cases are additive, it is concluded that the booster launch window is approximately one minute. Some additional capability may be possible by considering a variable rendezvous velocity; however, it is felt that the launch window would remain rather small, because deceleration is relatively large at these speeds.

## 7.0 RENDEZVOUS GUIDANCE REQUIREMENTS

A study was made to determine rendezvous guidance characteristics and requirements for the case of orbiter-airplane rendezvous at subsonic speed. The primary result of the study is the definition of one approach to providing the rendezvous guidance.

There are large variations of flight conditions, relative motion, and guidance activities between orbiter reentry and completion of rendezvous. Table 6 identifies various phases or events that occur during rendezvous, and indicates approximate time increments for each activity. It is assumed that tentative rendezvous position, flight conditions, and time have been defined prior to orbiter reentry, based on position of the orbital plane relative to the landing site, weather conditions, relative times of airplane takeoff and orbiter retro, and possibly other considerations. Both vehicles initially control to arrive at the specified position and velocity at the designated time. The orbiter continues this activity through the period of communication blackout. Updated rendezvous conditions may be computed aboard the airplane based on radar tracking of the orbiter during blackout.

Rendezvous conditions for the orbiter are updated frequently or continuously after communication blackout has ended. The orbiter maneuvers to achieve these conditions until time to begin a transition maneuver to establish nominal glide characteristics for terminal rendezvous and docking. After this time, all rendezvous maneuvers are performed by the airplane.

Relative motion of the orbiter with respect to the airplane, based on computations described in Section 6.0, is shown in Figure 42. During the final three minutes, the airplane controls relative position and velocity, based on radar and visual inputs, to maintain an efficient vertical closure.

Conditions at two minutes-to-go, in terms of relative altitude and altitude-rate, are very similar to those for approach to a Lunar landing, as shown by Figure 43. It is doubtful that rendezvous could be completed in such a brief time without accurate information on these parameters. The Apollo Lunar Module uses a landing radar to measure altitude and altitude-rate, and controls attitude and thrust as functions of these inputs. A similar system is required to complete the rendezvous as quickly as possible to maximize initial altitude and time available for docking. The rendezvous phase is completed at an altitude of 32,000 feet, with the orbiter above and within fifty feet of the airplane, and with relative velocity less than five feet per second.

For towing recovery, the guidance procedure is identical until the final minute, during which time the airplane slows to allow the orbiter to descend to a point below and ahead of the airplane to establish satisfactory conditions for towing acquisition.

## TABLE 6

## RENDEZVOUS PHASES

| PHASE | TIME-TO-GO, MIN. | DESCRIPTION |
|---|---|---|
| I | $>13$ | Early Reentry, both orbiter and airplane on inertial guidance. |
| II | 13-8 | Acquisition and update after blackout, continued guidance by orbiter. |
| III | 8-4 | Orbiter transition to $(L/D)_{MAX}$ and zero bank angle. |
| IV | 4-3 | Continued guidance of airplane by computer and sensors. |
| V | 3 | Begin visual guidance of airplane. |
| VI | 1 | Airplane pushover to match orbiter glide path. |
| VII | 1 - 0 | Approach to docking (or towing) contact. |

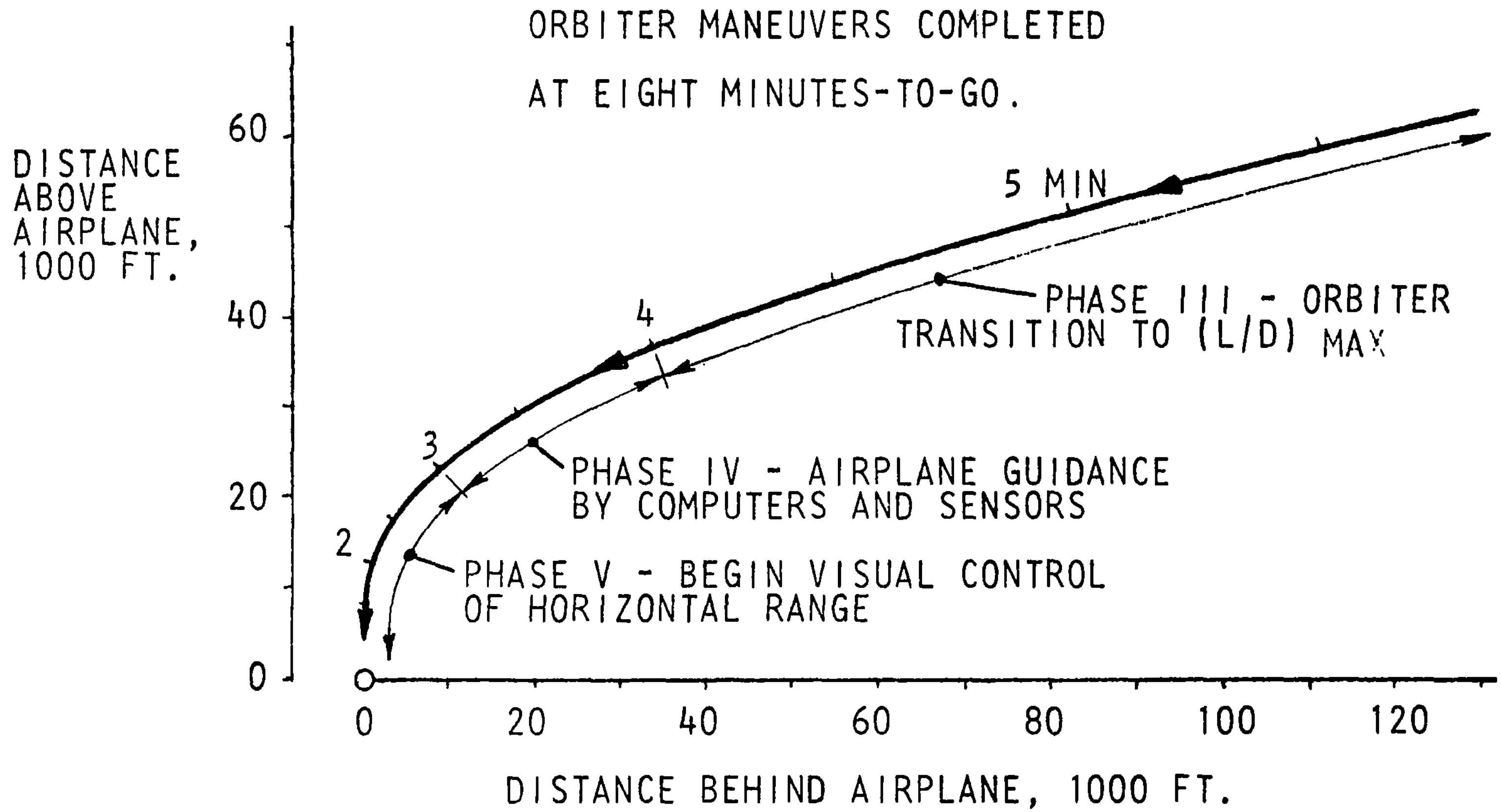



Figure 42. - Orbiter Motion Relative to Airplane (Five to Two Minutes-to-Go)

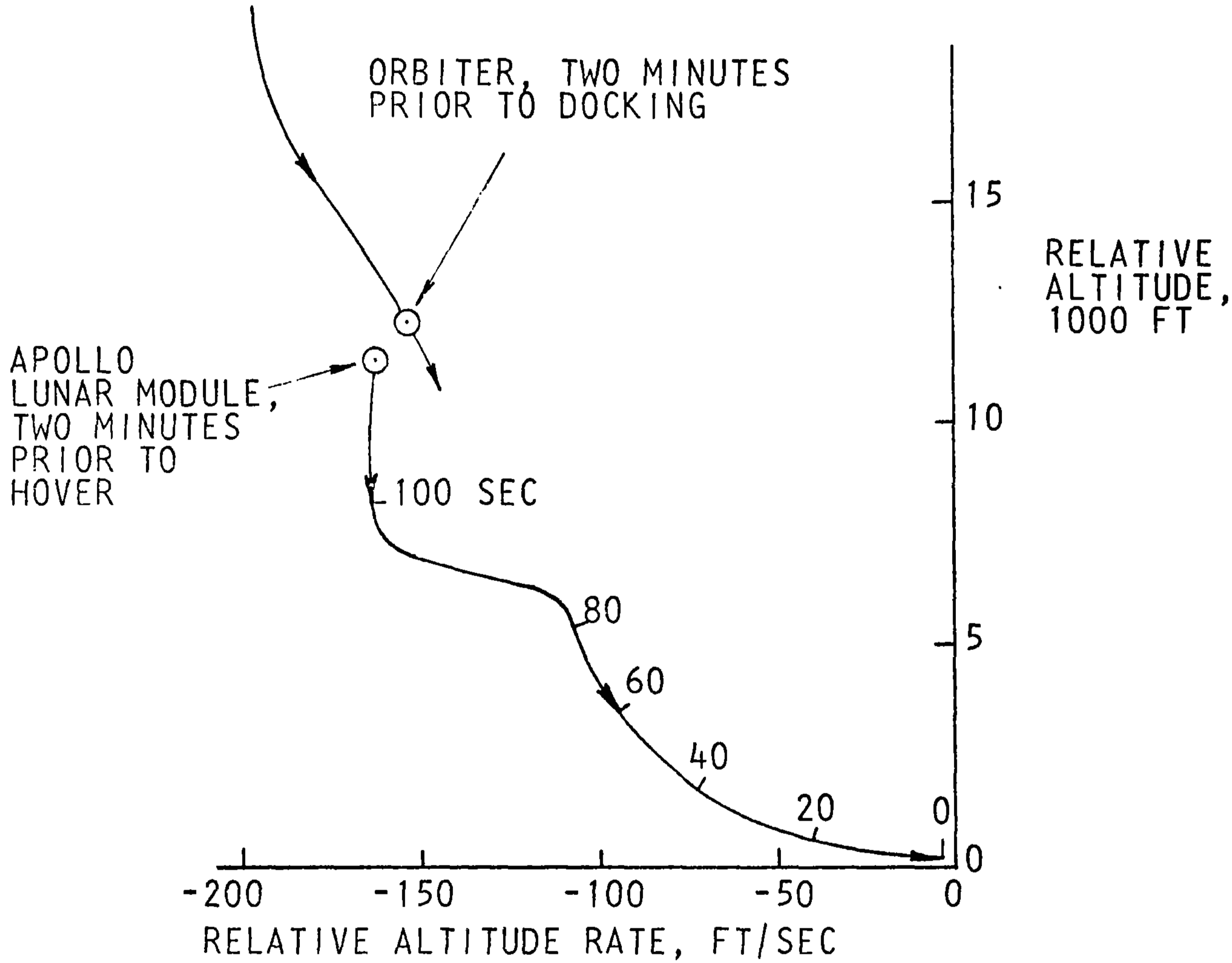



Figure 43. - Control of Relative Altitude

Rendezvous guidance requirements can be satisfied with inertial navigation, communication, and radar equipment. The radar equipment in the orbiter could include a transmitter, but may consist of only a transponder with relative motion data being provided by the airplane and communication equipment.

## 8.0 CONCLUSIONS

1. Atmospheric rendezvous is feasible.

2. General benefits of atmospheric rendezvous are:

   (1) A significantly smaller orbiter (approximately 20% reduction in weight).

   (2) Powered landings with go-around capability for all missions.

   (3) Lateral range capability that exceeds requirements.

3. General requirements for implementation of atmospheric rendezvous are:

   (1) Development of docking or towing techniques.

   (2) Design, manufacture, and qualification of docking or towing equipment.

   (3) A new or highly modified large airplane for recovery of the orbiter.

4. The orbiter-airplane rendezvous is considered superior to the orbiter-booster rendezvous for the following reasons:

   (1) Decreases in booster size and required launch fuel result from orbiter-airplane rendezvous. Booster size increases for orbiter-booster rendezvous due to increased wing area and cruise propulsion.

   (2) Design of a docking system is more difficult for the orbiter-booster case because of a more severe thermal environment.

   (3) Additional booster propulsion or orbiter drag control is required for a hypersonic orbiter-booster docking.

   (4) The booster must have additional cruise range capability after an orbiter-booster rendezvous, or it must be permitted to land downrange from the launch site.

   (5) The booster launch sindow is only about one minute for an orbiter-booster rendezvous.

5. Recovery of the booster with an airplane can result in a still smaller booster. Towing of the booster by an airplane can be

accomplished if additional airplane power is provided. Airplane-booster docking is not feasible unless an aircraft is used that is larger than the C-5A.

6. Rendezvous guidance requirements can be satisfied for each vehicle with an inertial guidance platform, computer, and radar and communication equipment. Visual observations are also required for terminal rendezvous and docking.

7. Time available for orbiter-airplane docking or towing is satisfactory.

8. There are no serious thermal problems for design of the orbiter-airplane docking or towing equipment.

9. Conceptual designs defined in this report for recovery by docking and towing are both attractive approaches to performing the recovery operation, and both are believed to include the basic features necessary for solution of design problems.

## 9.0 FURTHER STUDIES

Several additional studies are required for development, analysis, and evaluation of an atmospheric rendezvous capability. Preliminary design data for vehicles and docking and towing devices are needed to provide a basis for analysis of atmospheric rendezvous in more detail, including a detailed weight analysis to provide a quantitative evaluation of the benefit of atmospheric rendezvous from the standpoint of total vehicle weight. An analysis of the dynamics of docking is required, including all significant modes of motion, aerodynamic disturbances, boom flexibility, and dynamics of control inputs. At the completion of this work, there should be sufficient data available to compare and select either the docking or towing approach for further development.

A rendezvous guidance technique must be defined in detail and analyzed for stability and accuracy. Manned simulation studies can then be made to evaluate and define procedures for rendezvous and for docking or performing a towed landing.

Studies are also necessary to determine the effect of atmospheric rendezvous on crew safety and abort procedures.

Analysis is required for comparison of atmospheric rendezvous with more conventional recovery methods from a cost effectiveness standpoint.

Table 7 presents a list of these recommended studies.

## TABLE 7

## RECOMMENDED FUTURE STUDIES

1. Preliminary design of orbiter
2. Preliminary design of docking and towing devices
3. Definition of airplane modification and preliminary design
4. Dynamic analysis of docking operation
5. Analysis and development of rendezvous guidance technique
6. Rendezvous simulation
7. Simulation of towed landing or docking
8. Crew safety and abort studies
9. Cost effectiveness studies

# APPENDIX A

## RECOVERY CONCEPTS

Table A-1 lists recovery concepts that were considered early in this study. Those selected for study in more detail were:

(1) Docking of orbiter and booster (Concept No. 1 in Table A-1).

(2) Docking of orbiter and airplane (Concept No. 2).

(3) Towing of the orbiter by the airplane (Concept No. 3).

These are discussed in the main body of this report. The others were considered less attractive, and were studied only to the extent required to identify the advantages and disadvantages listed in Table A-1.

TABLE A-1

ORBITER RECOVERY CONCEPTS

| CONCEPT | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 1. Dock and land with recoverable booster. | (1) Integrated booster-orbiter attachments for launch and docking. No additional vehicle required.<br>(2) More efficient utilization of booster, i.e., booster responsible for 2 orbiters per mission.<br>(3) Fairly long time period available to achieve docking.<br>(4) General comment - applies to rendezvous concept in general - wingless orbiter may be sufficiently sound aerodynamically & structurally to accomplish emergency landing on foam-covered runway or water in the event of rendezvous failure. | (1) Large booster size for landing with orbiter aboard.<br>(2) Heating problems for hypersonic docking.<br>(3) Additional booster aerodynamic and/or propulsion requirements for docking.<br>(4) High risk, expensive testing for hypersonic docking.<br>(5) Requires 1 or 2 additional crew members and remote crew stations for booster.<br>(6) Concept not applicable for expendable boosters.<br>(7) Possible maneuverability problems with 2 unpowered vehicles.<br>(8) Booster angle-of-attack considerably larger than orbiter.<br>(9) Booster in non-equilibrium glide much of time.<br>(10) Increased booster cruise range, or land downrange from launch site. |
| 2. Dock and land with Airplane. | (1) Lower risk, powered, airplane landing.<br>(2) Heating problems less than for hypersonic docking with booster<br>(3) Powered aircraft able to make any necessary altitude & speed adjustments prior to docking<br>(4) Special airplane, but can be used for other purposes. | (1) Extensive airplane modification or new airplane.<br>(2) Extensive development for docking mechanism.<br>(3) Aircraft may require additional power plants. |

TABLE A-1 (Cont.)

| CONCEPT | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 3. Tow by airplane, land while towed during low-altitude fly-by. | (1) Less airplane modification than for hard docking.<br>(2) Land 10 to 20 knots slower than orbiter in free flight.<br>(3) Minimum heating problems.<br>(4) Several passes may be possible if initial hook-up fails.<br>(5) Minimum aerodynamic interference problems.<br>(6) Can be used for other purposes. | (1) High weight of cable and winch.<br>(2) Landing gear required for orbiter, or exotic prepared landing site.<br>(3) Special airplane; may require additional engines. |
| 4. Tow by booster. | (1) No additional vehicle required.<br>(2) Possibly smaller booster than for Concept No. 1.<br>(3) Efficient utilization of boosters. | (1) Aerodynamic heating of tow cable.<br>(2) Not applicable for expendable boosters.<br>(3) Increased cruise range or land downrange from launch site.<br>(4) Towing capability poor compared to airplane (See Section 4.2.3). |
| 5. Transfer of landing package from airplane to orbiter (wing, landing gear, possibly engines and fuel). | (1) Orbiter has high quality landing system without taking it into orbit.<br>(2) Ferry capability<br>(3) Wave-off capability<br>(4) May not require additional aircraft engines. | (1) Large aerodynamic interference forces during docking of orbiter and landing package. |
| 6. Airplane tow, circling letdown. | (1) Low-speed landing. | (1) Technique believed to be not applicable to large payloads.<br>(2) Impact gear. |
| 7. Tow by airplane, descend by parachute. | (1) Relative simplicity. | (1) Poor control of impact point.<br>(2) Shock absorption system required. |

TABLE A-1 (Cont.)

| CONCEPT | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 8. Tow by airplane, land with parawing. | (1) Low-speed landing.<br>(2) Good landing accuracy. | (1) Development for Apollo Program incomplete.<br>(2) Application to 200,000 lb. payload may present additional problems.<br>(3) Landing gear required. |
| 9. Tow by airplane to barrier arrestment. | | (1) High speed, high load factor landing objectionable to passengers.<br>(2) Landing gear and barrier load points required.<br>(3) No alternate landing site. |
| 10. Tow by airplane, balloon station delivery. | (1) Minimum speed landing. | (1) Very large balloons required.<br>(2) Possibly high load factor during balloon arrestment.<br>(3) No alternate landing site. |
| 11. Helicopter landing | (1) Minimum speed landing. | (1) Existing helicoptors limited to much smaller payloads. Largest helicopter has only 88,000 lb. capability. |
| 12. Sea landing | (1) Reduced accuracy requirement for landing.<br>(2) No landing distance requirement.<br>(3) Reduced lateral range requirement.<br>(4) No atmospheric rendezvous or docking required. | (1) Large naval support program required.<br>(2) Salt water environment requirement for all systems.<br>(3) Transportation back to launch site.<br>(4) Landing characteristics may be poor. |

TABLE A-1 (Cont.)

| CONCEPT | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 13. Dock and land with flying-wing type airplane. | (1) Special case of Concept No. 2. Structural and aerodynamic integration may be better. | (1) New large airplane. Unorthodox design. |

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